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INTERIM TECHNICAL REPORT

INVESTIGATION AND DEVELOPMENT OF NEW CONCEPTS FOR IMPROVEMENT OF AIRCRAFT ELECTRICAL POWER SYSTEMS

A REVIEW OF THE ELECTRICAL POWER SYSTEMS AND EQUIPMENT IN EXISTING COMMERCIAL AIRCRAFT

by C. H. Lee
J. J. Brandner

August 1969

Prepared under Contract No. NAS 12-659 by
AIRESEARCH MANUFACTURING COMPANY
Los Angeles, California

ELECTRONICS RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
CAMBRIDGE, MASSACHUSETTS 02139

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NASA ELECTRONICS RESEARCH CENTER
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PREFACE

The work in this interim report was performed by the AiResearch Manufacturing Company, Torrance, Calif., a division of The Garrett Corporation, under NASA Contract NAS12-659 for the Electronics Research Center, National Aeronautics and Space Administration, Cambridge, Mass. Dr. F. C. Schwarz and F. L. Raposa are the technical monitors for NASA-ERC. The work in this study program is divided into three phases:

(1) Review of present aircraft electrical systems and equipment, their performance characteristics, parametric data, capabilities, and limitations.

(2) Investigation of possible improvements of aircraft electrical power systems using state-of-the-art technology.

(3) Formulation of a philosophy for devising an optimum aircraft electrical power system based on technology extrapolated to the 1980's.

Documentation of the study program will also be divided into the three parts indicated above. This interim report presents the results of the first phase of the study and is a point of departure into the latter two phases.

This report was written by Dr. C. H. Lee (principal investigator) and J. J. Brandner. Other AiResearch personnel contributing to the program were K. M. Chirgwin, C. Y. Chin, J. H. Fu, Mac Hamilton, J. L. Lau, Jon Mandell, W. I. McAuliffe, and R. E. Vesque. The AiResearch document number for this report is 69-5193.

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INTRODUCTION AND SUMMARY

This is the interim technical report on a study entitled "Investigation and Development of New Concepts for Improvement of Aircraft Electrical Power Systems" funded by NASA Electronics Research Center, Cambridge, Massachusetts, Contract No. NAS12-659. Work under this contract was initiated on 14 May 1968. The original plan called for a two and one-half man-year level of effort for one year. After six months of study, work performed indicated that there was considerable incentive to deepen and broaden the scope of the investigation. With the approval of NASA Electronics Research Center, the study was extended one more year to July 1970 with an additional two and one quarter man-year level of effort. This interim technical report covers only information pertaining to the electrical power systems and components on existing commercial aircraft.

Objective and Method of Approach

The study objective is to formulate a philosophy for devising optimum electrical power systems for advanced aircraft. The philosophy recommended will consider improvements in reliability, safety, weight, size, and other factors that would result in increased revenue for transport aircraft; it will apply and extend advances in space power technology and related fields to aircraft electrical power systems. The study will review the entire aircraft electrical power system, including generation, conversion, distribution, and utilization equipment, in accordance with the above objectives, independent of traditional 400-Hz aircraft power technology.

Obtaining aircraft economy and reliability are complicated and involved tasks. The following factors have been considered in the search for improvements in the aircraft electrical systems:

- Simplicity--A simpler system will be more reliable and more economical.
- Automation--Additional automation will facilitate system manageability and increase aircraft safety.
- Weight--A reduction in the weight of the electrical system will increase revenue. For example, consider a typical large subsonic aircraft with payload capacity of 43,000 lb and electrical system weight of 11,000 lb. A 30-percent reduction in electrical system weight will increase the payload capacity appreciably by 8 percent.
- Protection--Better system protection will increase system safety.
- Information display--Better information display will also increase system safety.
- Fault prediction, detection, and isolation--These will reduce maintenance cost and increase reliability.

- Performance--Improved performance will result in higher efficiency and increased system capability.

- Components--Use of better components will result in higher system reliability and less maintenance.

The present aircraft electrical system (115/200-V, 3-phase, 400-Hz constant frequency ac) was developed about 25 years ago. This system concept was based on the needs at that time with relatively small aircraft. Many of the future aircraft will be much larger in size and will demand much more electrical power, part of which must be at very precise quality. Flying with higher speed at higher altitude, the environmental conditions for the future advanced aircraft will also be different from those of aircraft 25 years ago. Recent technical advances may also outdate some of the equipment and concepts presently used in aircraft. This study is therefore advisable to determine whether the long established electrical system configuration is still suitable for use in the future advanced aircraft.

The study is oriented primarily toward the relative merits of various power system concepts. To accomplish this, it is necessary to establish the approximate performance capabilities of the individual components so that they can be used collectively to indicate system performance. Consequently, the performance data in this report are representative rather than detailed component performance predictions.

The study information has been divided into three categories:

(1) Equipment in use whose performance primarily reflects technology existing at the time of its design (present large subsonic transports were designed in the late 1950's; the small subsonic transports in the early 1960's)

(2) Equipment that can be designed with current technology and has not yet been used in aircraft

(3) Equipment that can be predicted with technology anticipated for 1980 to 1985.

Meaningful presentation of the information from this study requires the application of anticipated component performance data to the various system configurations to arrive at relative system performance. A baseline system configuration typical of existing large subsonic aircraft will be used as a reference point with which to compare the candidate systems.

The anticipated output of this study program will be the documentation of a philosophy for devising optimum aircraft electrical systems and the selection and description of candidate configurations for future aircraft electrical power systems. The performance (weight, reliability, size, maintainability, safety, etc.) of the candidate systems will be compared. In addition, significant technological problems applicable to selection of the most promising candidate system will be discussed.

Scope of Investigation

The scope of investigation is defined by the following study tasks:

(1) Conduct an industry and literature survey on present day practice and state of the art in aircraft electric power generation, transformation, distribution, and utilization including present and projected electric loads on aircraft.

(2) Optimize the methods to supply electric power to the various loads to suit their inherent functional mechanism.

(3) Compile the necessary parametric information for the power system components on physical size, weight, and performance for the given power levels vs power system configuration, voltage, and frequency. This information will be based on the present state of the art of system component technology.

(4) Investigate application of new materials such as exotic magnetic materials, new insulation materials, unconventional conductor materials, etc for critical electrical system components.

(5) Investigate new system component concepts for application to aircraft in the sense of the objectives of this program.

(6) Review and analyze the preferable power system heat transfer technique.

(7) Specify the techniques and displays required such as failure prediction, detection and compensation, efficient energy management, and data monitoring for the acquisition and characterization of the electric power system.

(8) Investigate power system component reliability and its tradeoff against component cost and weight.

(9) Investigate component maintainability and its tradeoff against component cost and weight.

(10) Survey and determine the static and dynamic power requirements for present hydraulic and pneumatic powered loads and establish the tradeoff criteria in the selection and application of electric or combined electric-hydraulic and electric-pneumatic power subsystems for that purpose.

(11) Determine the relative weight of factors that determine the "effectiveness" of the electric power system in the sense of program objectives.

(12) Formulate a philosophy of aircraft electric power system design consistent with the objectives stated above.

(13) Devise and analyze power system candidates with inclusion of system design and optimization. Provide complete power system diagrams and descriptions.

(14) Identify the significant research and technology problems associated with achieving the most promising candidate electric power systems.

Progress in The First Year

During the first year, existing literature was surveyed to establish the state of the art and trends in system design. A bibliography on aircraft electrical power systems and components was compiled. Trips were made to major aircraft manufacturers, users, hardware suppliers, and government agencies to obtain data and operating experience on the electric power systems of existing and planned aircraft. These data were needed to establish configuration and design requirements, and to evaluate typical current electrical power systems.

More than 60 people in responsible positions in various organizations were visited all having considerable experience and background in the engineering discipline of aircraft electrical systems. Generally persons contacted recognized the need and the timing of the study and appreciated the farsightedness of NASA in initiating such a program. Hope was expressed that the result of this program should be fruitful and beneficial to the aircraft electrical industry.

The following activities were performed during the first year.

(1) The work on electrical load analysis began with a classification of the electrical loads, and a weight analysis of the constant and wild frequency ac motors, and brushless dc motors. The parametric data of aircraft utilization components are divided into three groups: (a) Components in existing aircraft, (b) Current technology components to be applied to aircraft, and (c) Components with future technology. The utilization weights were compared as a function of system frequency (including dc).

(2) In the area of power distribution, design considerations on safety, corona, voltage, and frequency levels were investigated. Characteristics and performance data on cables, relays, circuit breakers, contactors, and transformer-rectifier units were collected. Data for voltage and frequency effect on wiring weight were prepared.

(3) Power conversion techniques were considered and compared. Parametric weight data were obtained for the various possible techniques. Selection of an optimum conversion system cannot be made without system optimization since the type of conversion equipment is dependent upon the load location and type, and the generating equipment output.

(4) Possible methods of generating electric power were examined. The established existing generation system is the constant speed, constant frequency (CSCF) system. Current technology systems under development are variable speed, constant frequency (VSCF), rotating inverter (which uses two rotating machines to provide the required power), and high-voltage dc systems. Analysis indicates that the more advanced generating concepts can offer performance advantages in comparison to the present CSCF systems while still remaining weight competitive. The cycloconverter VSCF system, tentatively selected for use on the SST, appears to be particularly attractive; another possibility is the Learverter system, although information on it is limited. Parametric information on emergency power sources was also collected.

(5) The design criteria and performance capabilities of both the hydraulic and pneumatic systems have been surveyed because many aircraft loads are presently serviced by hydraulic or penumatic power. The survey indicated that the major hydraulic loads are flight controls, landing gear, brakes, and thrust reversers. It also indicated that for reasonable distances between the power source and the load, a hydraulic system can provide the load power for a lower weight than can the electric system.

(6) System control and protection methods in existing aircraft were reviewed and various protection schemes for single and parallel system operations were analyzed.

(7) Present techniques of cooling aircraft electrical components and heat transfer systems were studied. Possible improvements in heat transfer methods utilizing state-of-the-art technology were also considered.

(8) The objectives and criteria of aircraft electrical system reliability were outlined. The level of reliability currently achieved by various aircraft components and subsystem was also tabulated.

As mentioned previously, only that information pertaining to equipment in existing aircraft is presented in this interim technical report.

Work for The Second Year

Work for the second year will include the performance of study tasks (2), (7), (8), (9), (11), (12), (13), (14), listed above, and the continuation of tasks (1), (3), (4), (5), and (10).

Report Summary

This report is divided into eight sections; a brief summary of each section is given below.

System description and performance--This section expalins the configurations and performance requirements of aircraft electrical power systems. A representative system is then discussed in detail with respect to its control and protection functions.

Utilization equipment--Information on electrical loads and parametric data on utilization components on existing aircraft are presented. The different methods of performing load functions (mechanical, hydraulic, pneumatic, and electrical) are compared.

Distribution subsystem--A typical aircraft distribution system in widespread use is discussed in detail in this section. Basic considerations in designing distribution systems as well as component characteristics are also presented.

Generation subsystem--In this section, generation schemes, hydraulic constant speed drive, and ac generators are described. Application data on emergency power sources are presented.

Reliability--The reliability objective and power system channel capacity are discussed. Typical reliability data of some aircraft components are also tabulated.

Heat transfer techniques--A short summary of the present techniques of cooling aircraft electrical components and aircraft heat transfer systems is presented in this section.

Capability constraints of existing aircraft electrical power systems--This section briefly outlines the limitations of the electrical power system in existing aircraft.

Survey of present technology improvement activities--A brief summary of present activites to improve the aircraft electrical power systems is presented here.

Six appendixes are included in this report.

Power characteristics and system component requirements--This is an abstract of a design specification for a typical aircraft electrical power system.

Typical SST airplane electrical load data--Typical SST electrical data are presented for the purpose of comparison with subsonic aircraft electrical system data.

Aircraft hydraulic systems and equipment--This is a summary of the hydraulic systems and equipment on existing aircraft or aircraft presently under construction.

Heat transfer methods on existing aircraft--A description, including sketches and photographs, of the heat transfer methods being used for aircraft electrical equipment is presented in this appendix.

References--This is a list of references given in this report.

Bibliography--A selected bibliography pertaining to the technology of aircraft electrical systems is listed here.

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SYSTEM DESCRIPTION AND PERFORMANCE

Introduction

An aircraft electrical power system can generally be divided into three segments: generation, distribution and utilization. A schematic diagram of such a system is shown in fig. 1.

The input power requirements and component locations of the utilization equipment determine the distribution and generation subsystems needed. The interface of the utilization with the distribution equipment can be expressed in the following variables:

Load locations

Individual load power requirement

Load profile

Individual load power quality

In turn, the interface of the distribution equipment with the generation equipment is as follows:

Total load demand

Power type (dc or ac at what frequency)

Voltage

Power quality

Electrical power systems for aircraft application have become more specialized and unique as each new airframe imposes its own combination of requirements. A survey of 400-Hz ac systems in use today would disclose very few units which have rating, speed, type of cooling, and configuration all in common. Much of this diversification is due to the numerous types of aircraft and flight objectives. Some differences arise because special emphasis has been placed on certain design parameters such as reliability or cost.

The design goal for commercial aircraft, in general, is an optimum combination of dispatch reliability, system weight, and utilization time. The requirement of incentive for maximum return on invested dollar has led to the use of economic penalty values in design. Even for identical aircraft, these values will vary considerably, according to the method the manufacturer used to obtain data. For large subsonic transports, approximately 1 lb of dead weight is equivalent to about $160, and a flight delay of 1 hr represents a loss of revenue as high as $5000. For supersonic transports, these figures are about twice as high. With this economic consideration, the electrical utilization equipment would be composed ideally of simple, reliable, and durable designs of relatively light weight.

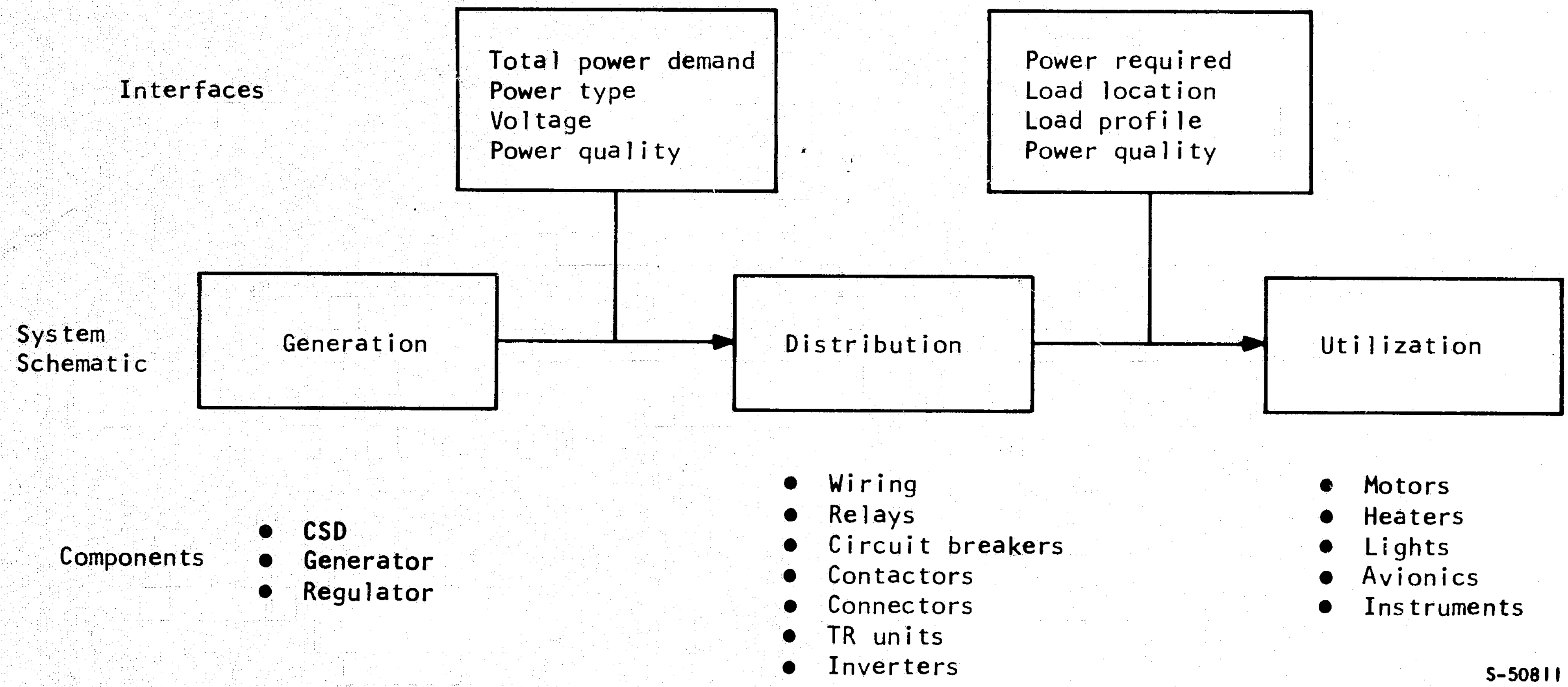


Figure 1. Aircraft Electrical Power System Schematic

To approach the program objective for the development of new concept in the aircraft electrical field, a thorough knowledge of existing systems, their requirements and their present status of technology is mandatory. In conjunction with the acquisition of this knowledge, an extensive literature survey and consultation of cognizant people in the aircraft industry have been undertaken as part of the program activity.

An outline illustrating the design requirements of a typical aircraft electrical power system is given in Appendix A.

Typical Aircraft Electrical Power Systems

Existing electric power systems on large transport aircraft can typically supply up to about 240 kVA of power. The total loads on the system at any time rarely exceed about 60 percent of the system capacity; the remaining power is provided for redundancy, degraded operating modes, and to satisfy inrush currents for motors. Approximately 95 percent of the demanded power is used in the ac form (with about 5 percent of the ac power input to avionics being converted to dc power inside the avionic packages). The remaining 5 percent is supplied as dc power in some avionic equipment, controls, relays, solenoids, lamps, and heaters. The total weight of a typical electric power system on a large, existing transport aircraft is about 11,000 lb (5,000 kg). This weight can be divided into the major categories of (1) generating equipment for power supply, (2) distribution equipment to control and carry the power to the loads, and (3) utilization equipment for power consumption. The total weight includes the weight of the driven load (fan, blower, pump, etc.) as well as the weight of the supporting structure when it is changed with changes in its driving motor. An analysis has indicated that the total weight of 11,000 lb (5,000 kg) is apportioned within the power system as follows:

(1) <u>Generating equipment</u>--6 percent of the total, or about 700 lb (318 kg)

(2) <u>Distribution equipment</u>--30 percent of the total, or about 3,300 lb (1,500 kg)

(3) <u>Utilization equipment</u>--64 percent of the total, or about 7,000 lb (3,180 kg)

The system has four different modes of operation which are defined as follows:

(1) <u>Normal flight operation</u>--Generators driven by aircraft engines supply all loads in the aircraft.

(2) <u>Ground power operation</u>--Power for lighting, air conditioning, other passenger comfort loads, and equipment warmup supplied to the aircraft from a ground power cart or onboard auxiliary power unit.

(3) <u>Degraded flight operation</u>--In a four or three channel system, it is designed to be dispatchable with a single generator or power channel inoperative; with two of the four channels inoperative, almost all of the loads can still be supplied but passenger discomfort may result; with three of the four, or two of the three channels inoperative, the aircraft would have to descend to the lower altitude, but would still have all of the vital aircraft loads supplied.

(4) Emergency flight operation--With all channels inoperative, or only a single channel operative, vital aircraft flight controls and navigation equipment would be supplied from an emergency inverter and a battery with about 50 A-hr capacity.

Power is generated by the aircraft engines on 2 to 4 identical channels (depending on the number of aircraft engines), each about 20 to 60 kVA. Hydraulic constant speed drive units are used to accommodate the approximately 2 to 1 output speed ratio of the engine. The generators are brushless, rotating rectifier machines that weight (generator only, CSD not included) about 1.0 lb/kVA (.45 kg/kVA) output in the 60 kVA rating on new second generation jets and about 2.5 lb/kVA (1.2 kg/kVA) output in the 60 kVA rating on older, established aircraft. Typically, the older design generators use air cooling with auxiliary power for excitation and operate at 6000 rpm while the new designs use oil cooling with a permanent magnet generator (PMG) on the same shaft for excitation and operate at 8000 rpm.

The output from the generators is 120/208-V, 3-phase grounded neutral, 400-Hz power and conforms to MIL-STD-704. Generally, the quality of all of the power is determined by the most exacting of the individual load requirements. This power is typically transmitted over a distance of as much as 150 ft (46 m) to the main power distribution center located near the flight engineer's station. The distribution center ties together all the generators as well as an attachment for ground power input when the aircraft engines are not operating. Bus tie breakers and generator circuit breakers allow the generators to be operated together in parallel (as is normally the case) or as independent power sources. The paralleling can be done automatically by sensing the voltage and phase relationship between each channel and the paralleling bus and closing the bus tie breaker when both voltages and frequencies are within prescribed tolerances. Controls and instrumentation are provided on the flight engineer's panel to manually override and control the paralleling or isolated operation of the generators. Appropriate sensing circuits are incorporated in the system to ensure equal real and reactive load division between generators.

Protection incorporated in the electric power system includes differential protection for the generators and feeders, overvoltage, undervoltage, overfrequency, underfrequency, underspeed on the generators, overexcitation and underexcitation for the paralleling operation, and current imbalance on the paralleling bus. The ground power connection incorporates phase sequencing to prevent ground power from being incorrectly connected.

From the generator load bus, several lines carry the power throughout the aircraft to localized distribution buses which transmit the power to the individual loads. Each of these lines is connected to both the paralleling bus and to one of the generator feeder lines. The connection is between the generator circuit breaker and the bus tie circuit breaker. Consequently, the loads can be serviced from either the paralleling bus or the generator (if it is being operated in isolation from the paralleling bus).

A nominal 28 Vdc power system is provided to supply those loads requiring dc power. The dc power is supplied to the dc loads during normal operation by a number of transformer rectifier (TR) units. There are usually as many TR units

as generating channels. The rating of these units is typically 50A. However, ratings up to 200A are not uncommon. Each TR unit is connected to a generator load bus and powers an associated 28 Vdc load bus. Normally, the units operate in parallel; however, a blocking diode prevents the essential TR from feeding other than the essential dc bus. In addition to the TR units, a 24-V battery provides a standby source. This source supplies backup electrical system control power and can supply power to minimum navigation, instrument, and communication systems should all other sources fail.

In addition to a constant frequency system, some aircraft installations include a variable frequency system. Because a great portion of the electrical loads are of pure resistive nature (de-icing, galley, and incandescent lighting), the variable frequency characteristic of the power source will have no adverse effect upon the operation of such loads. The variable frequency generating system is simplified because it does not employ a constant speed drive unit. Reliability and efficiency is, therefore, substantially better; there is one power link less involved in producing the output. This advantage is offset, however, since two different types of distribution systems with associated control and protection are necessary. Variable frequency distribution is also less economical than constant frequency distribution.

System Description and Performance

Generating system.--A typical present-day electrical generating system, which is of the constant-speed, constant frequency (CSCF) type, that may consist of either three or four generating channels operating in parallel is discussed herein. Fig 2 illustrates a block diagram of the basic ac system. Each generator is driven by a hydraulic constant speed drive (CSD), which, in turn, receives its power from the accessory drive pad of one of the main jet engines. Driving torque to the generator is supplied at a constant speed of 6000 (8-pole generator), 8000 (6-pole generator), or 12,000 rpm (4-pole generator), providing a constant frequency output of 400 Hz. Each generating channel is provided with voltage regulation, control, and protection equipment. The principles of operation of these equipment will be discussed later in this section under the headings of system control and system protection.

CSD input speed variation is typically 2 to 1. The three-phase (four-wire grounded neutral) power output from the generator is carried over a typical 60- to 120-ft (18 to 37 m) transmission line to the main power shield, located near the flight engineer's station in the control cabin. The power shield is the collection point for the feeders of all generating channels. It contains the individual generator circuit breakers (GCB), the bus tie breakers (BTB), the generator load buses, and the synchronizing bus. The power from the generator load buses is distributed via thermal breakers to the various local distribution buses, and from there to the various loads throughout the airplane. Each generator can operate isolated, supplying its own load bus; or the generators can be paralleled in any desired combination by closing the proper bus tie breakers. Any load bus can obtain power from other generator on the system by opening of the proper GCB and closing of the proper BTB. Also located in the power shield and connected to the synchronizing bus is the external power contactor (EPC). During ground servicing operation, power can be supplied to the synchronizing

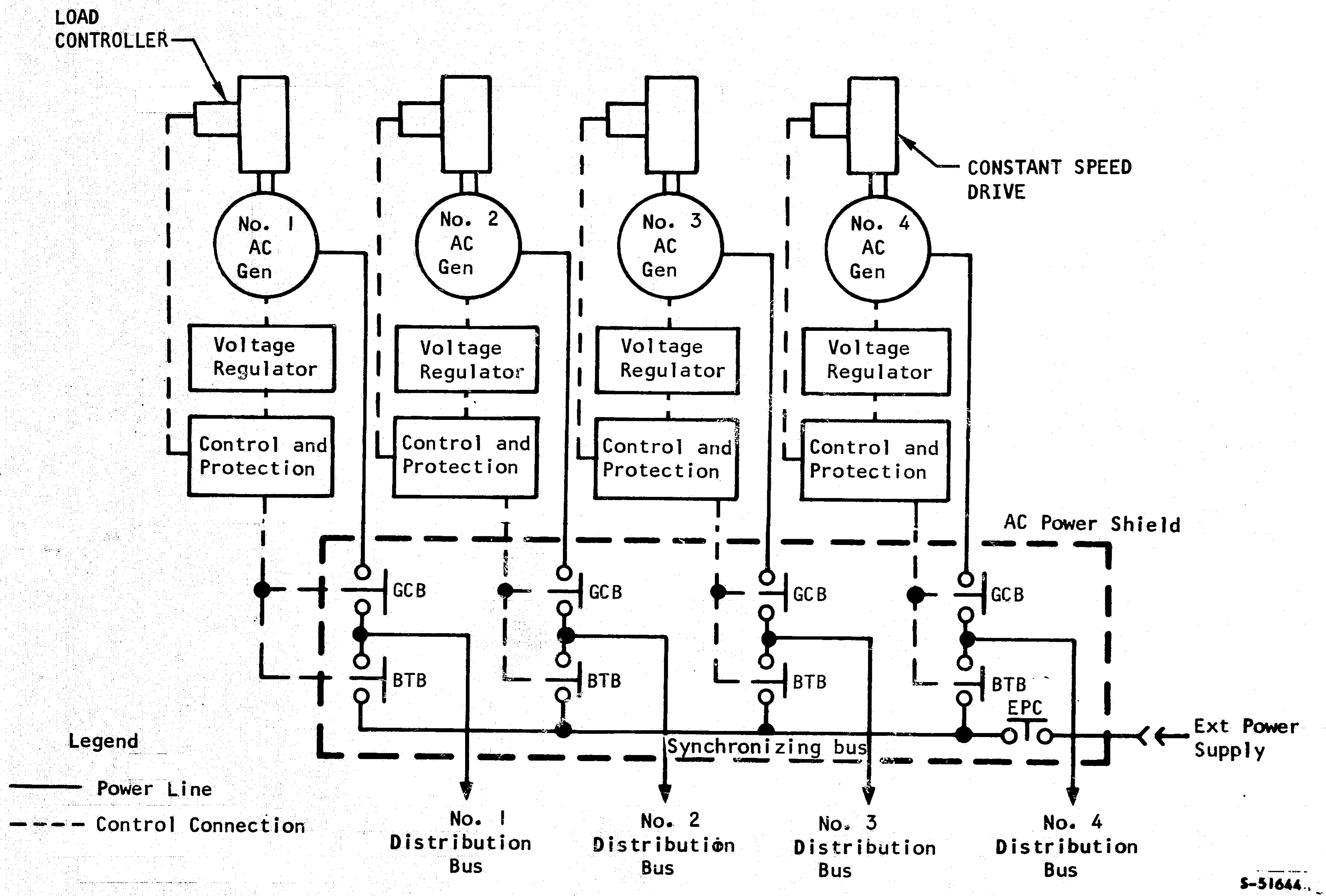


Figure 2. Typical Four Generator Bus System

bus and, hence, to the load buses from an auxiliary ground power unit. Some electrical systems incorporate an airplane-installed auxiliary power unit (APU) which might also be electrically connected to the synchronizing bus (not shown in fig. 2).

The main ac power shield serves as the point of regulation for each generating system where nominal system voltage of 115/200 V is maintained. Design criteria for the performance, control, and protection of the system include the following:

(1) Continuous supply to load buses (during normal and most abnormal operation of the system, power shall be maintained to the load buses).

(2) No single fault shall cause loss of power to an unfaulted load bus.

(3) Maintain system capacity (during normal operation, load sharing circuits shall provide maximum system capacity and shall remove only that faulted generating channel which might endanger the overall system or load equipment).

(4) Maximum protection with minimum components.

In order to provide the necessary degree of system control and to achieve system reliability, each power source, mechanical and electrical, requires controlling, regulating, and protective equipment.

Fig. 3 shows a block diagram of a single generating system with control and protection tie points. The CT package and potential transformer provide real and reactive load sharing signals to modify the output speed of the CSD and the generator excitation through the load controller and voltage regulator respectively. The differential CT's detect the internal faults of the generator. The control and protection block include many functions. The purposes of this group of equipment are load control, autoparalleling and fault isolation. In case of a fault on the synchronizing bus (except balanced 3 phase fault which is unlikely) the negative sequence relay will provide a trip signal to all bus tie breakers, thus permitting each generator to continue supplying its load bus independently. The external power contactor EPC is controlled by a phase sequence relay, the negative sequence relay, the external power switch, and a ground power switch on the flight engineer's panel. The EPC is interlocked with all GCB's such that closing the external power contractor will trip all GCB's.

<u>Constant speed drive</u>.--The alternator drive is a governed hydraulic transmission which transforms the varying input speed from the aircraft engine to constant speed for driving the alternator. Hydromechanical constant speed drives are composed of basic hydraulic components which can be arranged in various combinations to meet specific performance requirements. The components are generally a variable hydraulic displacement pump and a fixed hydraulic displacement motor. The pumping unit volumetric displacement is varied in relation to input speed to provide an oil flow to the fixed displacement motor unit which will assure a constant output speed. An output speed signal, converted to a hydraulic signal by the governor, actuates a servo piston which is connected to a pump wobbler plate, thereby varying the displacement of the pump.

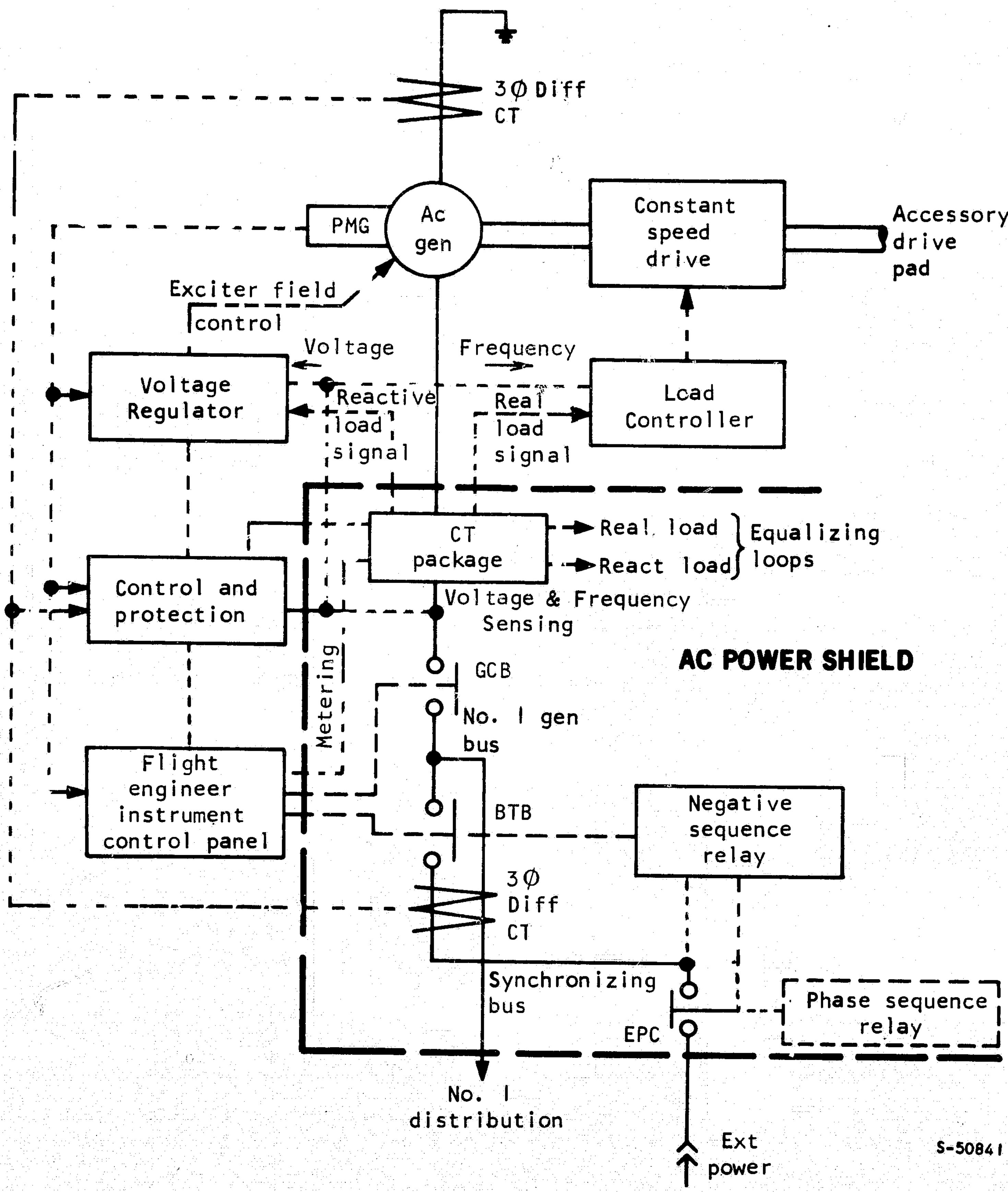


Figure 3. Generator Control and Protection Block Diagram

The drive provides a constant output frequency of 400 ±1 Hz for steady-state conditions of load and input speed variations (typically 4000 to 8000 rpm). With rapid acceleration or deceleration of input speed or normal changes in load on the aircraft electrical system, transient output frequency is held to 400 ±4 Hz. Under steady-state conditions of load and speed when two or more alternators driven by constant speed drives are operated in parallel, the drive output frequency is held to 400 ±1 Hz, and real load division is held within 2 kW (typical for alternators rated 40 kVA each). With rapid acceleration and deceleration of input speed, or sudden changes in load on the aircraft electrical system, real load division is held within 12 kW total.

Alternator.--The alternator with the typical ratings of 30 kVA, 40 kVA, and for more recent application, 60 kVA, is generally of the brushless rotating rectifier type with salient pole construction, and is either air or oil cooled. On the newer systems the ac main generator incorporates a small pilot permanent magnet generator (PMG) for system self sufficiency. The PMG output (typically 500 VA) serves for exciter excitation and for system control. Alternator speeds are 6000 rpm, 8000 rpm, and, for recent applications, 12,000 rpm.

System Control

When operated singly, the two main controlling functions for each system are frequency regulation (by trimming the output speed of the constant speed drive) and voltage regulation (by varying generator excitation). These two control functions change to real load division control and reactive load division control respectively when the systems operate in parallel.

Frequency control.--A block diagram of the control system is shown in fig. 4.

Regardless of the particular trim control method that is used, the basic building blocks of the electrical trim will always include the following elements:

(1) Frequency reference

(2) Converter or comparator to obtain a voltage or current analog of frequency error

(3) Amplifier

(4) Transducer to modify the speed governor setting

The frequency reference is the standard to which the generator frequency is continuously compared. A voltage or current analog of the frequency error is obtained from the comparator. This frequency error signal is then amplified and applied to the magnetic trim governor of the trasmission. Current in the trim coil functions as vernier control of the governor. In order to distribute the real load equally between generators, the direction and magnitude of the deviation of each generator load from the average load is sensed by the load division loop. This loop is composed of a series string of parallel combinations of current transformer secondaries and burden resistors (see fig. 5). Each current transformer (CT) is on the same phase of each generator.

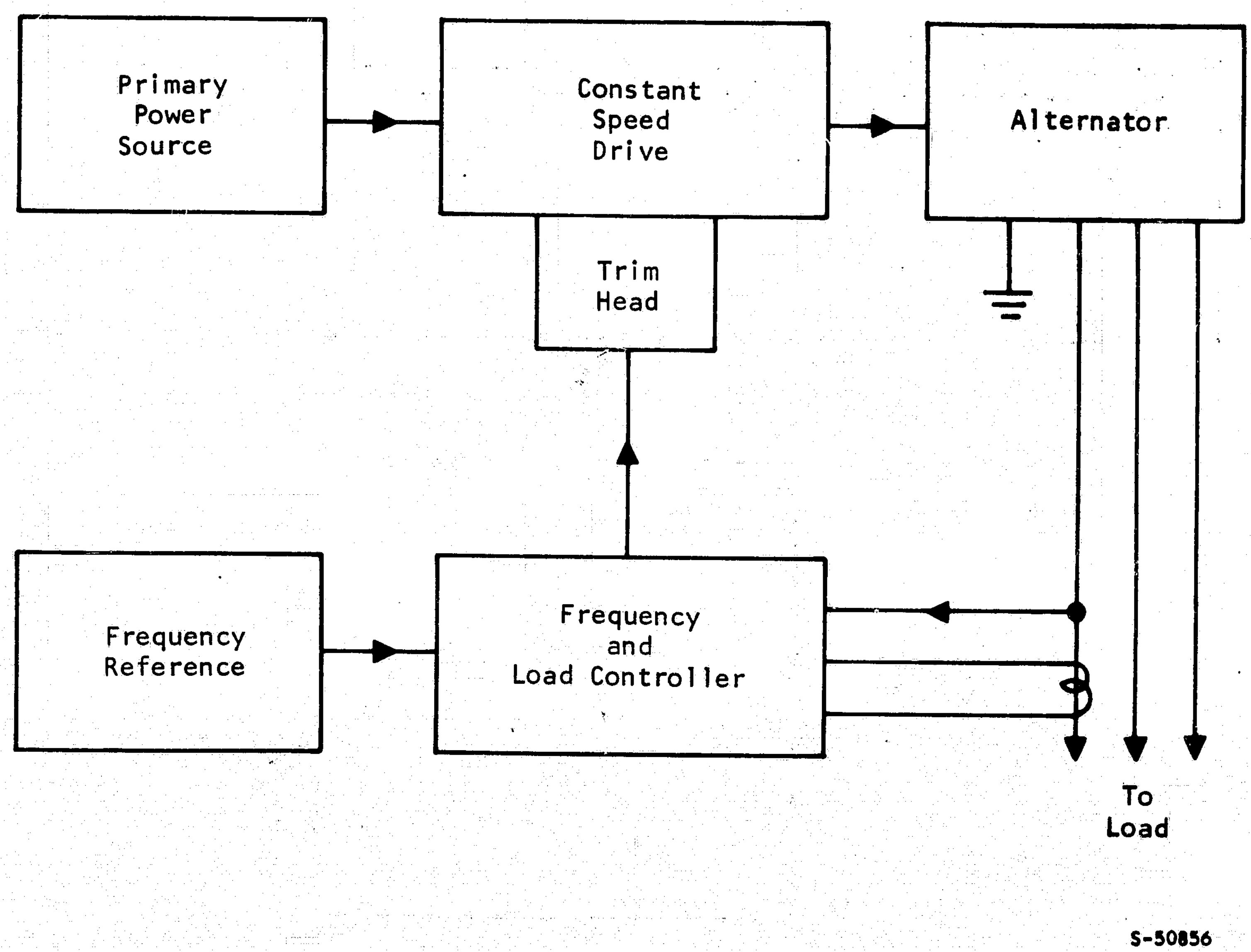
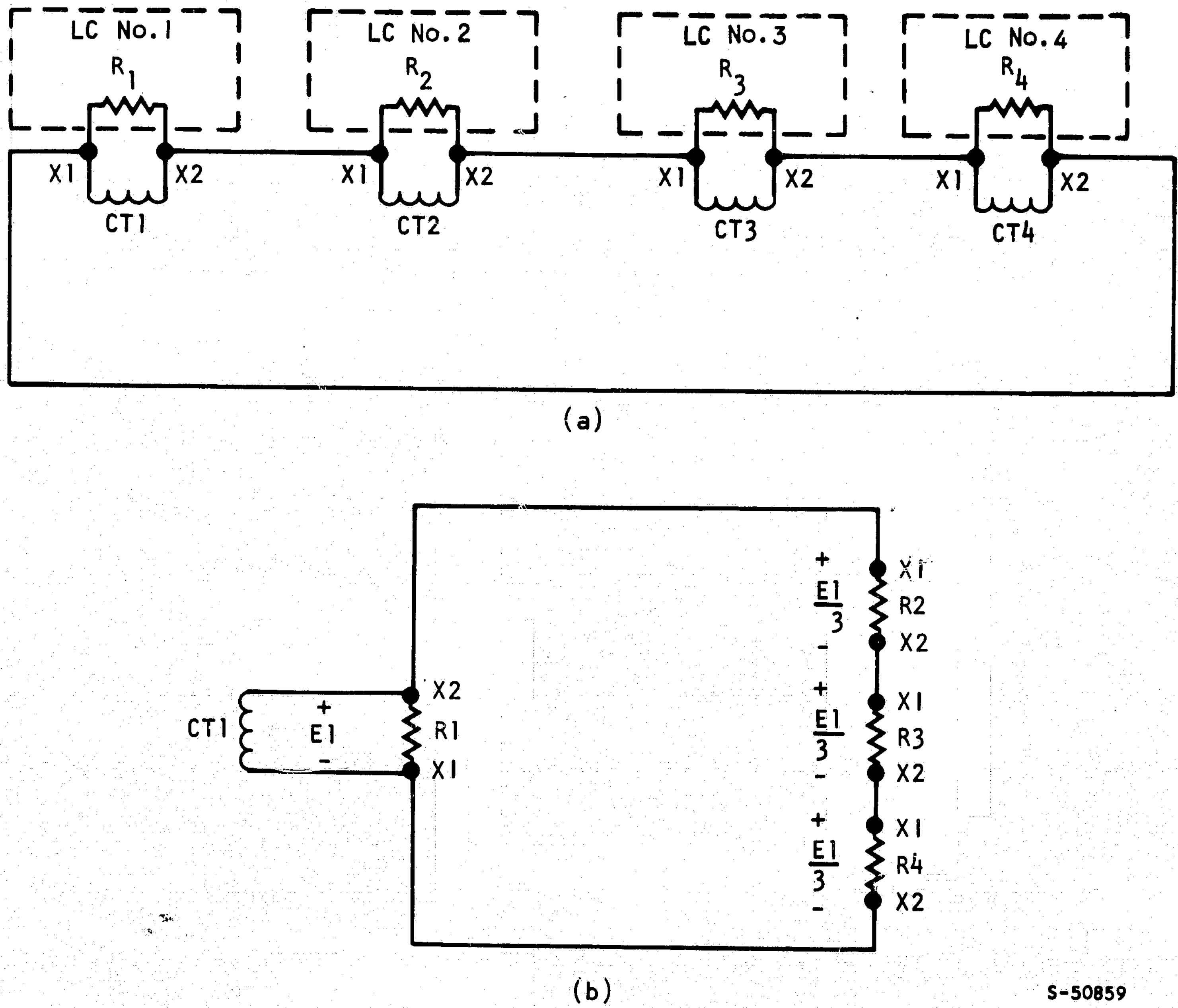


Figure 4. Frequency and Real Load Sharing Control



LC = Load Controller

Figure 5. Load Division Loop and Voltage Distribution

An example will best illustrate the operation of the load division loop. Suppose that a voltage El is produced in CTI as a result of the induced current, and that E2 is induced in CT2, E3 in CT3, and E4 in CT4. Consider El, E2, E3, and E4 instantaneous voltages, the induced voltage at one particular instant of time. It is apparent that El will appear across Rl. El is shown as a voltage drop across Rl with an arbitrarily assigned polarity. Fig. 5(b) shows the burden resistors with the current transformers removed. The transformers are removed so that the voltage El distribution among the resistors can be seen more clearly. Since El is dropped across Rl, it can be seen that El will also be dropped across the series combination of R2, R3, and R4. Since all of these resistors have the same value, one-third of El will be dropped across each of them.

Note that the polarity of the voltage dropped by each resistor is opposite to the polarity of E2 across R2, E3 across R3, and E4 across R4. In a similar manner, E2 will be dropped across R2, while one-third of E2 will appear across each of the other resistors, but of the opposite polarity with respect to E2. Likewise, E3 and E4 will be distributed among the resistors. As a result of voltages El, E2, E3, and E4 being induced in the load division loop, each resistor drops four separate voltages. On Rl the four voltages are El, one-third E2, one-third E3, and one-third E4 with El opposing the other three voltages. The net voltage dropped across Rl will be the algebraic sum of these four voltages. The net voltage dropped across R2 will be E2, one-third El, one-third E3, and one-third E4, and so on for R3 and R4. The net voltage across each burden resistor is the input to the load control circuit from the load division loop. Consider first the case where the load is balanced. The output of all four generators will be the same and El, E2, E3, and E4 will be equal. The net voltage dropped across each resistor will be zero, and there will be no output from the load controllers.

Now consider the case where the load is unbalanced. An unbalanced load condition exists when two or more generators are carrying more or less than their share of the total load. The share that each generator should normally carry is equal to the total load divided by the number of generators. In the load division loop, this share is represented by the sum of El, E2, E3, and E4 divided by four. In the unbalanced load condition the net voltage drop across Rl is still the algebraic sum of El, one-third E2, one-third E3 and one-third E4.

Since the voltages induced in the CT's are not necessarily equal, the resulting net voltage drop across Rl may have a certain magnitude and polarity. The magnitude indicates how much the output of generator no. l has deviated from its required share of the load, and the polarity indicates whether the generator is carrying more or less than its share of the load.

$$\text{Voltage drop across } R_1 = E_1 - \frac{E_2}{3} - \frac{E_3}{3} - \frac{E_4}{3}$$

$$= \left(I_1 - \frac{I_2}{3} - \frac{I_3}{3} - \frac{I_4}{3}\right) R$$

$$= \left[\left(\frac{4}{3} I_1 - \frac{I_1}{3}\right) - \frac{I_2}{3} - \frac{I_3}{3} - \frac{I_4}{3}\right] R$$

$$= \frac{4R}{3}\left[I_1 - \frac{I_1 + I_2 + I_3 + I_4}{4}\right]$$

$$= \frac{4R}{3}\left[I_1 - \text{average } \bar{I}\right]$$

Similar expressions can be obtained for voltage drops across R_2, R_3 and R_4.

Since the current sensed by the current transformers in the same generator phases represents total or apparent current, reference to the in-phase component with the voltage has to be made within the load controller to obtain the correct real load division. Reference of the in-phase component of the current may be obtained by a phase sensitive demodulator as shown in fig. 6. The dc output of the modulator is proportional to the magnitude of the product VI cos θ, where θ is the phase angle between line current and phase voltage. In fig. 6a, the primary of the potential transformer T_1 is connected across one phase of the generator. The two secondaries are connected to two diode bridges in opposite polarities as shown by the polarity dots in the figure. The current derived from the current transformer will flow in either one of the two secondaries depending on the instantaneous relative polarities of the line current and phase voltage. Fig. 6b shows a case that line current I_1 lags behind the phase voltage V_{1-n} by a phase angle θ. The instantaneous value of i_o (fig. 6a) will be negative from 0 to t_1 and from t_2 to t_3; while it will be positive from t_1 to t_2 and from t_3 to t_4. The output of the demodulator, i.e., average value of i_o, is then

$$i_{o,ave} = \frac{1}{\pi}\int_{-\theta}^{\pi-\theta} I_{om} \sin x dx = \frac{2}{\pi} I_{om} \cos \theta$$

$$(I_{om} = \text{peak value of } i_o)$$

In other words, the output of the demodulator is proportional to the real component of the line current. Since phase voltage is a constant, the output of the demodulator can also be considered as proportional to VI cos θ.

It is important that the primary of the modulation transformer is connected to the same phase from which the current is sampled to obtain real load division. If a generating channel does not operate in parallel with the other channels, its corresponding BTB or GCB is open. This closes a normally open contact on the breaker, thereby shorting the output of the current transformer in the non-contributing system.

<u>Voltage regulation</u>.--Except for weight and size, the functional requirements of the voltage regulator have not changed over the years. These requirements are listed below.

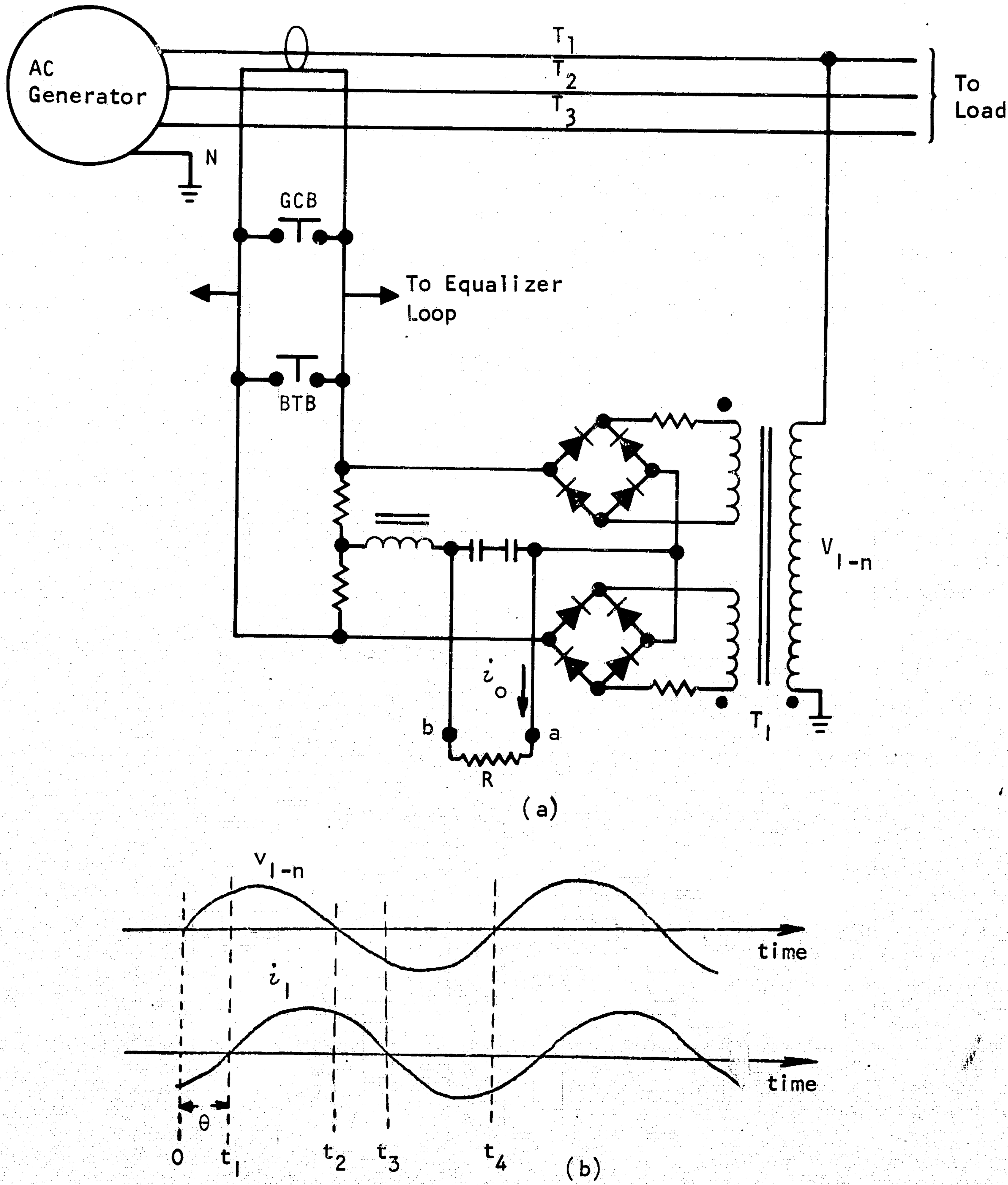


Figure 6. Phase Sensitive Demodulator

(1) Compensation for the high no-load to full-load regulation that is characteristic of unregulated ac generators with fixed excitation.

(2) Compensation for generator voltage change with change in machine speed.

(3) Compensation for the tendency of generator voltage to drift as the field winding of the machine heats up (resistance change).

(4) Rapid return of generator voltage to normal after transient changes in load occur, i.e. sudden application and removal of one to two per unit load, clearing of faults, etc.

(5) In parallel systems, control of the division of reactive loads supplied by each generator.

Although the functions remained the same, the means for accomplishing them changed as the technology progressed, resulting in some significant improvements. At one time, the standard regulator designed for use with rotating exciter machines was the carbon-pile regulator. Gradually, this type was replaced by the magnetic-amplifier regulator. In later applications, a voltage regulator comprised a transistor amplifier controlling a magnetic amplifier output stage was adopted. Present-day regulators, however, are completely transistorized, including the output stage.

Magnetic amplifier and transistorized regulators each have their own advantages and disadvantages. Magamp regulators are advantageous because they can withstand high temperature environments and are unsusceptible to electrical noises. Transistorized regulators are fast response, small size, and light weight.

Transistor regulators usually are switching type regulators and employ the pulse width modulation scheme. Fig. 7 shows the block diagram of a typical regulator. The three-phase, line-to-line voltage is sensed at the point of regulation and is stepped down by a three-phase transformer in the regulator. After half-wave rectification, the ripple voltage is subjected to a wave shaping network to produce a saw tooth waveform. This waveform is compared to a fixed, dc reference voltage. The rising or falling ramp of the saw tooth causes a varying pulse which is amplified and switches the output stage to control the current to the generator field. Power for exciter excitation and all control functions is obtained from the pilot PMG.

Equal reactive load division for each generator is provided by the reactive load equalizing loop. The principle is the same as that for real load sharing, except that the sine of the phase displacement angle between voltage and current (instead of the cosine) is used as factor to obtain the power product. This is done by connecting the primary of the modulation transformer (see fig. 6) between lines T2 and T3, those lines where the current sample is not taken as input to the modulator.

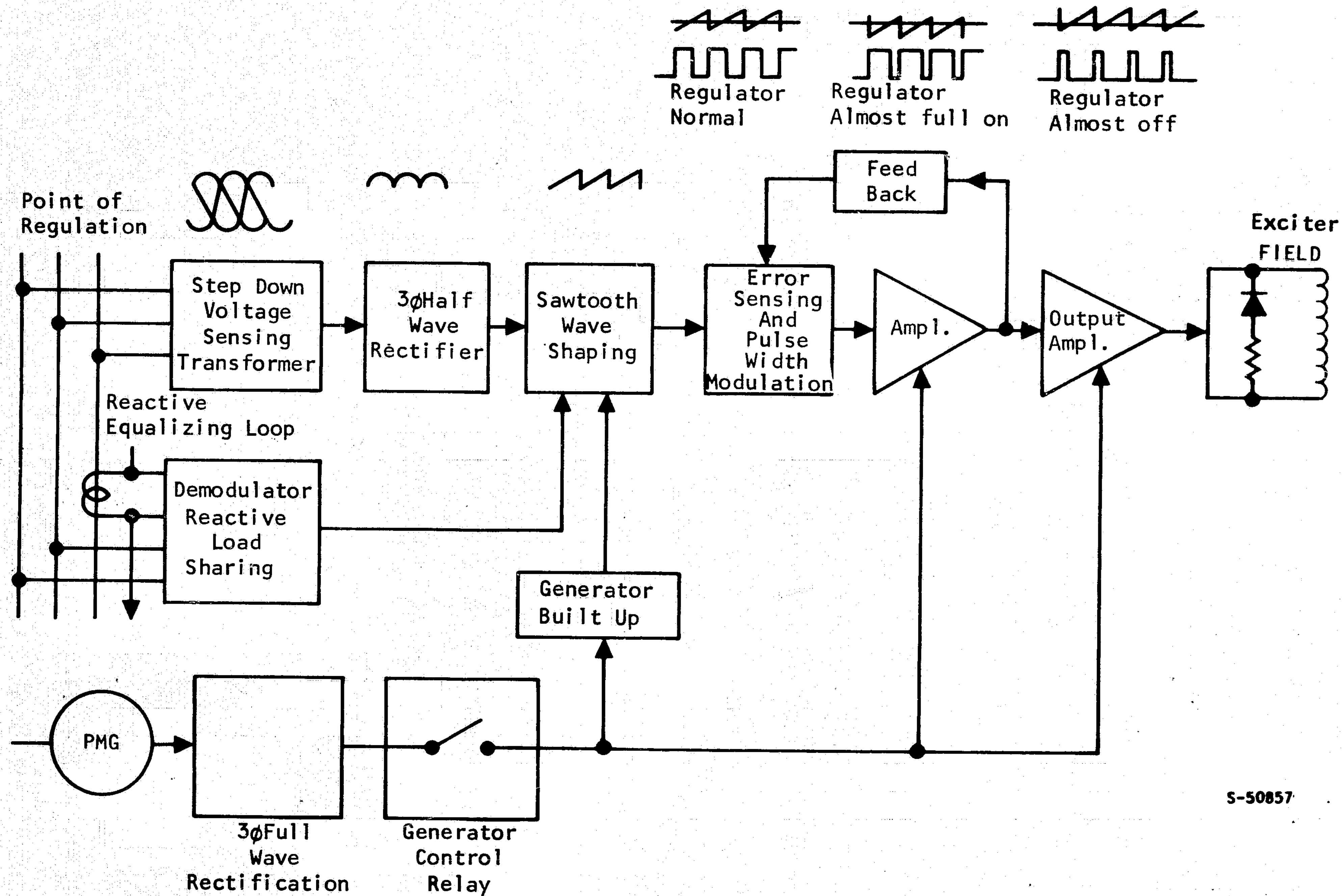


Figure 7. Basic Principle Operation of Transistorized Voltage Regulator

The resulting modulator output dc signal is added or subtracted to the reference in the pulse width modulator, depending whether the respective generator contributes less or more than its share of reactive load to the system.

Control panel.--The control panel provides system control and protection from abnormal operating conditions for each generating channel, the total generating plant, and the equipment load. These control, protective, and warning functions are listed below.

(1) Control functions

Generator field control

Generator circuit breaker control

Bus breaker control

Autoparallel

Time delay functions

(2) Protective functions (definitions of these functions are on pp. 26 and 28)

Overvoltage

Undervoltage

Overspeed, over frequency

Underspeed, under frequency

Overexcitation
Underexcitation } excessive reactive load division unbalance

Difference current protection

Differential protection

Open-phase protection

Synchronizing bus protection

External power out-of-tolerance protection

CSD protection

(3) Warning and emergency functions

Generator overtemperature

Emergency fire switch

This is believed to be a representative list of control and protective functions which could be provided by a supervisory panel for a conventional electrical system in contemporary aircraft. Existing control panels, however, do not provide all the protective functions listed above. Each individual application will call for an optimum number of protective functions influenced by the prevailing protection philosophy of the system designer. Some functions, such as inadequate external power quality and sync bus protection, which are usually not included in the generator control unit, are furnished by a separate bus protection panel.

A typical control and protection logic diagram is shown in fig. 8. (Warning functions are not shown in this diagram.)

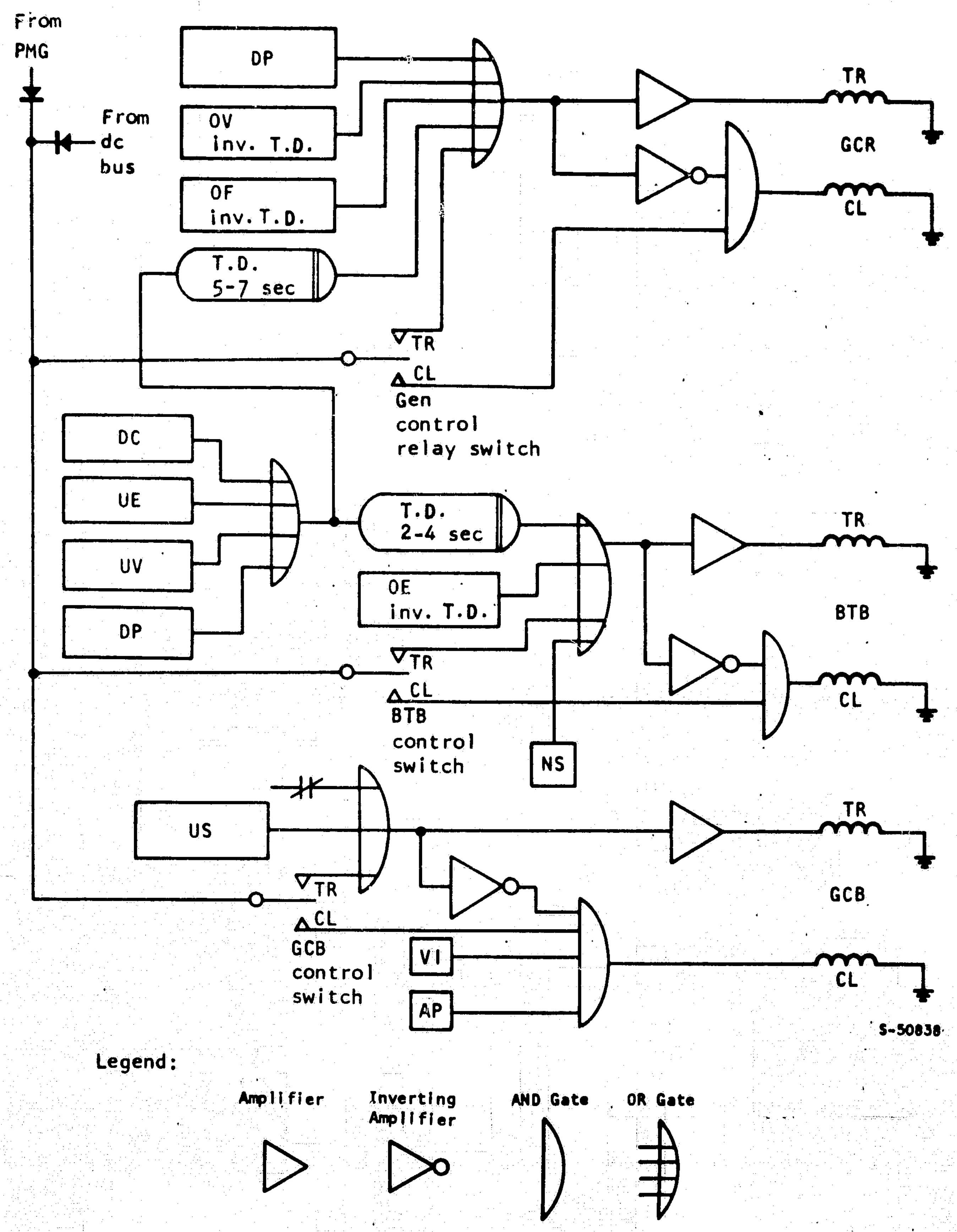
Control warning and emergency functions.--Control warning and emergency functions are discussed in the following paragraphs.

Generator control relay: The main function of the generator control relay is to open and close the exciter field current. It is generally a latch type relay with both a trip and a close coil, and with auxiliary contacts connected to the generator circuit breaker trip coil circuit. This causes the circuit breaker to open whenever the exciter field is opened. Relay position is usually indicated by a light to the flight engineer. Operation of the generator field relay may be semiautomatic (flight engineer may have a switch for manual control) or fully automatic (flight engineer has no control). In very recent control panels, this mechanical relay has been replaced by solid-state switches.

Generator circuit breaker (GCB) and bus tie breaker (BTB) control: These contactors are generally of the mechanically latching type. They require two separate input signals, one for closing and one for opening the breaker. These signals are furnished by small slave relays (mechanical or magnetic latch type) which control the main power contactors. Whenever a trip signal is present from a protective command, an inhibit signal prevents the respective circuit breaker from closing, thus providing anticycling as long as the abnormal condition persists. In very recent control panels, all mechanical relays have been replaced by solid-state switches; the main power contactors are the only mechanical switches in the system. The status of the GCB and BTB is indicated to the flight engineer by indicating lights.

Autoparallel: The automatic paralleling circuit operates to lockout the GCB closing circuit whenever the voltage frequency and phase angle are out of tolerance between the incoming generator and the synchronizing bus. Usually the circuit does not perform any corrective action to bring the two systems into synchronism. It merely prevents the incoming generator from being paralleled until all the conditions for paralleling are met. The circuit does not operate if the frequency difference exceeds 4 Hz or the phase angle difference is greater than 30 deg. If the frequency and phase angle requirements are satisfied, the circuit will operate with a voltage difference of 10 V or less line to neutral (L-N) between the two voltages.

Some systems are provided with synchronizing lights to permit and aid the operator in tying the system manually via the BTB if the autoparallel circuit malfunctions or during specific emergency procedures.



GCR	Generator control relay	TD	Time delay	OE	Over voltage excitation
BTB	Bus tie breaker	TR	Trip	NS	Negative sequence
GCB	Generator circuit breaker	CL	Close	OP	Open phase
DP	Differential protection	DC	Difference current	US	Under speed
OV	Over voltage	UE	Under excitation	VI	Voltage indication
OF	Over frequency	UV	Under voltage	AP	Auto parallel

Figure 8. Control and Protection Logic Diagram

Time delay functions: To discriminate the action of certain fault command signals, two different time delay circuits are simulataneously energized as shown in fig. 8. One time delay circuit actuates the BTB after 2 to 4 sec, thereby isolating that channel from the rest of the system. The other time delay circuit trips the GCR after 5 to 7 sec, which disconnects the generator from its load bus if the fault still persists. In this case, reclosure of the BTB is desired. On older systems, this is done manually; on more recent systems, it is done automatically.

Generator overtemperature: Some systems incorporate various warning functions; typical of these is generator overheat, which is indicated to the flight engineer by a warning light. The overheat warning could be caused by an overloaded generator, inadequate generator cooling, or generator mechanical failure. Whatever the cause may be, the overheat warning is announcing incipient generator failure, and that the generator should be deenergized and removed from its load bus. The practical value of this warning indication has been questioned by some airplane users in the past. If some warning requires a definite response by operating personnel, the corrective action could be done automatically as well.

Emergency fire: In a case of extreme emergency, where it would be desirable to completely deenergize the entire bus system, the flight engineer pushes the fire switch. This sends a signal to the 5- to 7-sec time delay and results in tripping of the GCR and GCB. Pushing the fire switch affects all generating channels simultaneously.

System Protection

Protective circuits defined.--The definition of the protective circuits are as follows:

(1) Over- and undervoltage protection--Prevents damage to load equipment in the event of excitation faults during single generator operation.

(2) Over- and underspeed (frequency) protection--Prevents improper operation of load equipment; it is usually included in the constant speed drive; it selects and isolates a faulted drive or an abnormal speed condition during engine idle or shutdown.

(3) Over- and underexcitation protection--Prevents damage to generating equipment in the event of excitation faults during parallel generator operation; selective isolation of the faulted system prevents power loss to the aircraft loads.

(4) Difference current protection--As for over- and underexcitation when channels operate in parallel, this protection selects and removes that generating channel which contributes significantly more or less than its share of total current to the loads.

(5) Differential fault protection--Guards against fault in the generating and transmission system only. In general, differential fault protection

covers all equipment located within the differential fault zone. Any applicable difference between current flowing out compared to the current flowing into the zone is detected and corrective action is initiated (line to line or line to ground faults).

(6) Open phase protection--Prevents damage to load and source equipment; if an open phase occurs anywhere in the system, and if there is danger of reducing system capacity, its respective system will be deenergized by operation of a zero sequence relay.

(7) Synchronizing bus protection--A distribution system protection may be obtained by a negative sequence relay on the synchronizing bus, thus providing unbalance voltage protection for that bus and tripping all BTB simultaneously in case a line to ground or line to line fault on the bus occurs.

(8) External power protection--To prevent improper operation and damage to load equipment, several protective functions already provided for the aircraft system may also be included for protection of the external power channels. This usually is protection against out of tolerance voltage and frequency. In case of excess deviation, the external power contactor (EPC) is tripped off. Phase sequence protection will prevent closure of the EPC should the ground power phase sequence be opposite to that in the aircraft.

(9) CSD protection--There are usually three protective devices incorporated in the constant speed drive. A shear section is provided on the input drive shaft for the protection of the main engine accessory gearbox. A solenoid operated emergency drive disconnect permits manual disconnect of the drive shaft in flight in case of generator or drive failure or overheat. The disconnect is controlled by a switch on the flight engineer's panel and cannot be reset in flight. Also, an overrunning clutch on the drive input prevents power flow in the reverse direction from the generator back into the drive in case of severe load unbalance or drive failure in parallel operation.

Normal system operation.--Normal system operation includes starting and shutdown. Normal system operation is depicted in the logic diagram of fig. 8 presented previously. When the aircraft engines are started and the generators are up to speed, sufficient output power for control and generator excitation is available from the PMG. The CSD, which uses the PMG output as referenced frequency, regulates the system to nominal output speed. If the previous shutdown of the system was normal and did not occur because of a fault, the generator control relay is in the close position and excitation current is automatically applied to the generator field. The voltage regulator will control the generator output voltage to 115 ±1 V at the point of regulation. Before the generator circuit breaker (GCB) can be closed, four conditions must be met for single generator operation.

(1) The generator must be up to speed, and no underspeed signal (US) present

(2) The generator control relay must be closed

(3) Proper generator output voltage must be indicated (VI)

(4) Sync bus voltage must be zero

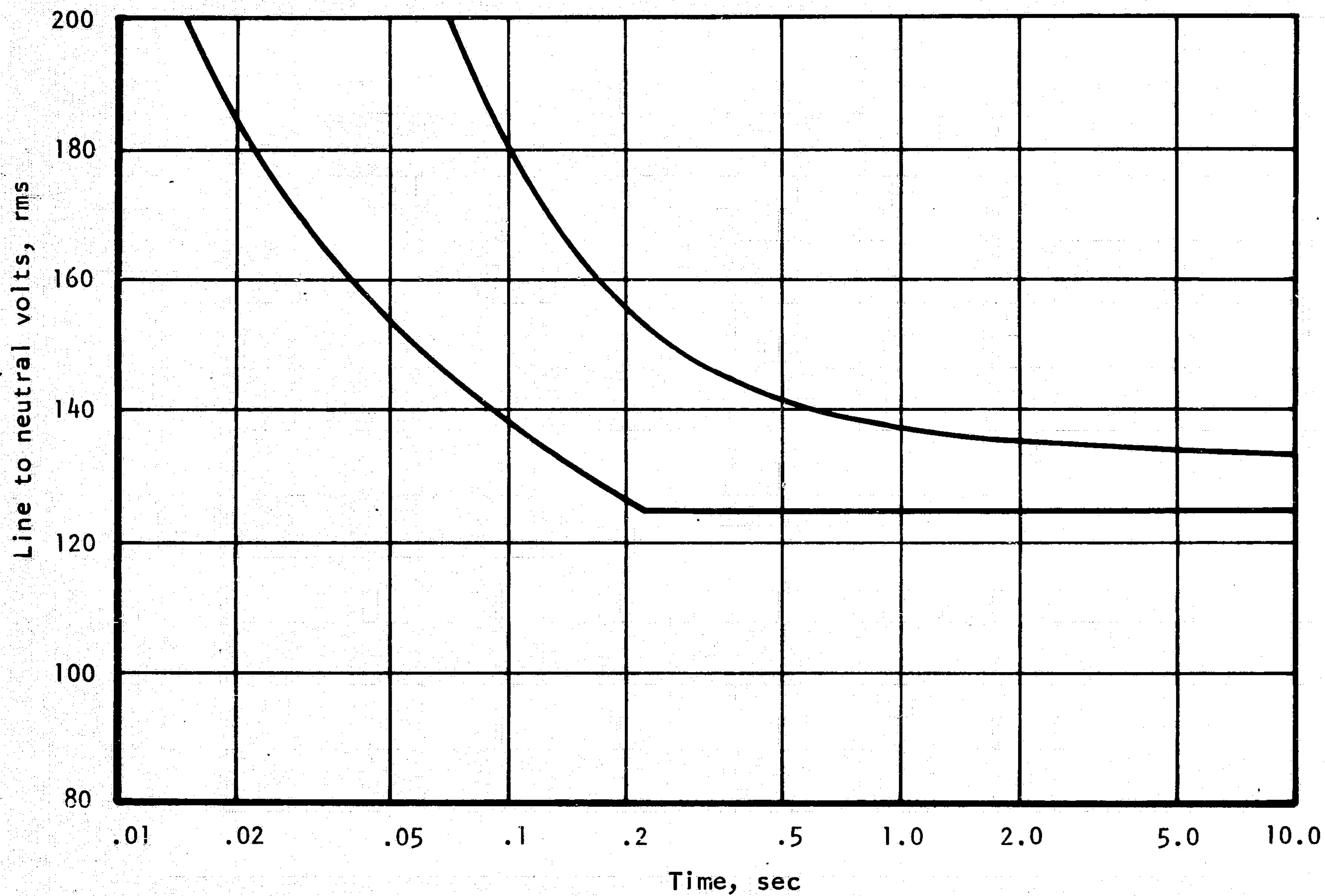
If all the aforementioned conditions are met, closing of the GCB switch on the flight engineer's panel will close the breaker, thus connecting the generator to its load bus. Normally all BTB are closed, and each oncoming channel after the first is paralleled via the GCB.

If the flight engineer wishes to isolate the load bus or synchronizing bus, the GCB switch and BTB switch can be moved to the trip position manually. If the aforementioned breakers are tripped, the associated generator may be deenergized by moving the GCR relay to the trip position. If the GCR and GCB are both closed, tripping the GCR will both deenergize the generator and automatically trip the GCB through the GCR contacts.

Normally, when no fault occurs, system shutdown is completely automatic. The generator circuit breakers are tripped by the underspeed circuit when the drive underspeed falls below 360 Hz. Thus, on shutdown the BTB and GCR both remain closed. Since the bus tie breakers are closed, all load buses can be energized by external power when it is connected to the switch. Also, since the generator control relay remains closed, the generators begin to build up automatically on startup as soon as control power is available from the PMG. Consequently, the only action required by the flight engineer on startup is to close the GCB switches.

<u>System protection for fault condition</u>.--System protection for fault condition is discussed for single generator and parallel systems.

Single generator system faults: Overvoltage protection is usually considered to be the most important single protective device in an aircraft. An overvoltage condition cannot only cause damage to load equipment(in particular, semiconductors) and decrease life expectancy, but it can also have adverse effect on the generating system (heating) and may lead to severe reduction in total system capacity. Overvoltage protection is designed to have an inverse time-voltage characteristic because much of the damage caused by overvoltage is due to overheating. To obtain overvoltage protection and at the same time prevent nuisance operations on load switching transients, the circuit must operate very rapidly on extreme overvoltage and relatively slow on slight overvoltage. On completion of the time delay, the generator control relay trips, deenergizing the generator and, consequently, tripping the GCB. Fig. 9 shows the operating limits of MIL-E-7894 for the overvoltage circuit. The overvoltage circuit senses the average of the three-phase generator output voltages and converts it to a proportional dc signal. This signal charges an appropriate RC timing network, which in turn initiates the trip signal to the GCR. Due to the similarity of the overvoltage and overexcitation functions, these circuits are usually combined.



S-50832

Figure 9. Operating Limits for Overvoltage Relay (per MIL-E-7894)

Undervoltage protection is needed to prevent damage to load equipment. Sufficient low voltage could result in high current and excessive I^2R losses in motor loads, causing overheating and eventual burn out. A condition of undervoltage exists whenever the system voltage falls below 100 V L-N. Load switching can result in momentary low voltage; however, time delays prevent nuisance operation by overriding these transients. The undervoltage is usually combined with the underexcitation circuit; its output starts different time delay circuits. In single generator operation, if an undervoltage condition exists for 5 to 7 sec the GCR will be tripped, deenergizing that generator and associated GCB. Overfrequency or underfrequency protection is provided in case of excessive drive speed deviation. It is generally initiated when the system frequency increases or decreases 10 percent from its nominal value. This will cause tripping of the GCR and GCB. The circuit operation may be based on the frequency to dc conversion, comparing the analog of the PMG frequency to a fixed reference, or it may utilize the digital principle comparing the period of the frequency to be sensed to a fixed time interval.

Differential protection is provided for protection of the power source and the primary transmission system only. This protection is required to respond instantly whenever the difference current within the differential protection zone exceeds approximately 30 percent of the systems per unit line current. Faults causing this difference current may be due to internal grounds in the generator or phase-to-phase or phase-to-ground faults within the protective zone. The differential protective loop extends from three current transformers connected on the neutral side of the generator to three current transformers connected on the synchronizing bus just beyond the bus tie breaker. A fault signal on the differential protection circuit will trip the GCR, deenergize the generator, and isolate the generator from its load bus by tripping the GCB. Some newer systems employ two or more differential protection loops, one for each generating channel and others to include the synchronizing bus with all its feeder lines from auxiliary and external power.

Open phase protection guards against excessive phase unbalance. It can be shown by symmetrical components that whenever an open phase occurs in a grounded neutral system of paralleled generators, zero sequence current will appear in the neutral connections. Thus, a properly designed sensing network in the neutral connection of the system will sense and detect the zero sequence component, and, if excessive, will initiate a time delay network. In single generator operation the system will be deenergized and removed from its load bus after the condition has persisted for 5 to 7 sec. Open phase sensing is usually incorporated into the differential protection circuit as a matter of convenience.

Parallel system faults: Parallel operation of a number of small generating units is a factor which contributes to the reliability of the service furnished by a power system. Maximum reliability is not attained when each unit supplies an isolated section of the total load as is necessary on a variable frequency generating system. Isolated operation results in a portion of the load being lost, at least momentarily, after failure of a generating unit and before the load may be divided among other machines.

In a parallel system, a generator can become overexcited or underexcited. But as it is in parallel with several other generators, the resultant voltage is the average of all the generator voltages and may not reach a value which would cause the overvoltage protection to operate. However, when a generator operating in parallel becomes overexcited, it tends to supply considerably more than its share of reactive power to the system. This condition causes the generator to become overloaded, and since this could cause the system capacity to be reduced, the faulty generator should be removed from the system. Overexcitation is any condition that produces excitation over and above normal exciter field current requirements Any overvoltage condition will also cause an overexcitation when the generators are operating in parallel. An open current transformer in the equalizer circuit and a shorted equalizer loop could cause the machine with the higher voltage to become overexcited. Overexcitation is overvoltage biased with a current signal and is sensed in exactly the same manner as the overvoltage circuit. Current is sensed by means of the equalizer loop. This provides selectivity and enables the system to select and shutdown the system causing the overexcitation.

Underexcitation is any condition which produces excitation below normal exciter field current requirements. When an underexcitation fault occurs on a generator operating in parallel with other generators, this generator becomes a load for the rest of the system and heavy reactive circulating currents flow between generators with resultant overheating. The undervoltage on the system due to the underexcitation fault on one generator is not as severe as it would be if only the one generator were operating by itself. When an underexcitation condition of a certain generator is detected (by means of the equalizer loop, same as the condition of overexcitation) two time delays are initiated. One will trip the BTB after about 2 to 4 sec. If the voltage of that isolated generator indeed shows an undervoltage, the second time delay will deenergize the GCR after another 1 to 3 sec.

Difference current protection is another precaution to ensure that equal load contribution by each generating channel is within tolerance. This protection senses the total current difference between paralleled generators and, after an inverse time delay, opens the BTB of a channel whose total current differs from the average by a predetermined amount.

Because the tie bus is not included in any of the protective schemes for each generating channel, there is usually a separate bus protection panel provided which protects not only the synchronizing bus but also the external power feeder lines and contactors.

On older systems, thermal protection for the synchronizing bus was provided, Some systems employ negative sequence voltage protection. This responds to line to ground faults and line to line faults, but will not respond to a three-phase symmetrical fault.

Recent protective schemes incorporate the synchronizing bus into a separate differential protection zone which includes auxiliary feeder lines and breakers. A differential fault within this zone will simultaneously trip all BTB and contactors connected to the zone.

Information Display

Higher performance aircraft and more complex avionics have given the pilot and flight engineer more work to do and less time to do it. More subsystem components and flight aids have caused cockpit readout congestion. As a result, the display of electrical power system information in current aircraft is rather simple. In a large commercial transport, the following information is usually displayed:

(1) Ac power

Phase voltage - A voltmeter with phase selection switch

Frequency

Power and reactive power - One single phase wattmeter per channel. By pushbutton action, this wattmeter can be changed to read reactive volt-amperes.

(2) Dc power

Voltage

Current

(3) Operational status

Generator circuit breaker, "on" or "off"

Bus tie circuit breaker, "on" or "off"

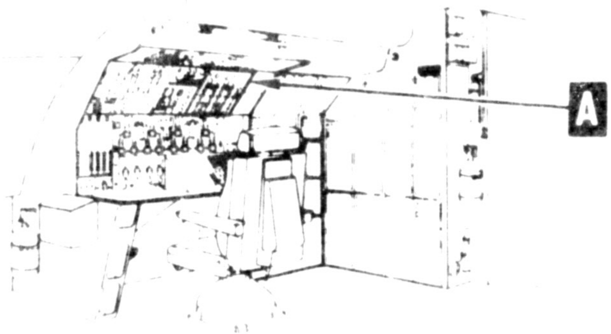
APU generator, "on" or "off"

External power contactor, "on" or "off"

In general, information will not be displayed unless the flight crew can use it to perform some corrective action during flight.

The ac system control panel in the C5-A airplane is shown in fig. 10. The C5-A represents the latest aircraft design.

The introduction of system and component failure prediction, detection, and compensation on future advanced aircraft will require a very different display system.



Flight engineer's station

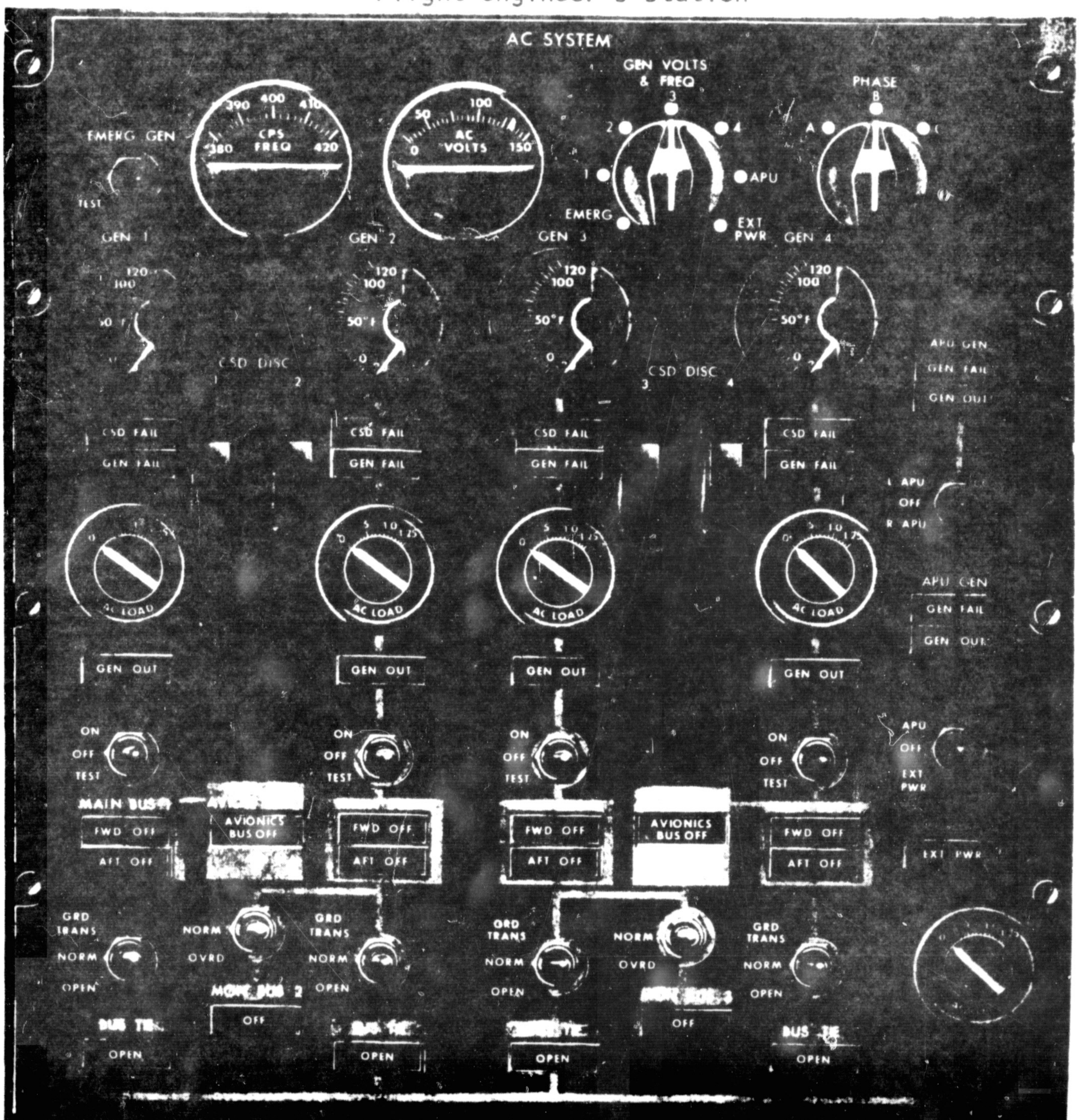


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Figure 10. Flight Engineer's AC System Control Panel

UTILIZATION EQUIPMENT

Introduction

Since utilization equipment represents a major portion of the aircraft electrical power system it warrants maximum design consideration. The achievement of a good electrical system design begins with a thorough knowledge of the aircraft load requirements, methods of supplying electrical power, and the characteristics of the components that perform the load functions. A collection of this information is presented in this section in addition to a general review of the utilization equipment currently installed in commercial aircraft.

An example of electrical utilization weights for a large subsonic transport are listed below. The weight of the electrical equipment includes the supporting structure and the pertinent auxiliaries. For a motor load, the weight includes the motor and the load it drives, if the weight of the load changes with motor design such as a change in motor speed.

Equipment	Weight
Flight control	380 lb (172 kg)
Starting system	122 lb (56 kg)
Fuel system	232 lb (105 kg)
Lighting	619 lb (280 kg)
De-icing and miscellaneous heating	275 lb (125 kg)
Air conditioning	235 lb (107 kg)
Avionics	1837 lb (833 kg)
Instrumentation	1268 lb (576 kg)
Fire protection	102 lb (46 kg)
Pneumatic power system	50 lb (23 kg)
Hydraulic system	133 lb (60 kg)
Galley and lavatories	1750 lb (793 kg)
Water and waste	64 lb (29 kg)
Total	7,067 lb (3,205 kg)

Load Distribution and Power Quality Requirements

Typical aircraft loads.-- Electrical loads in a commercial aircraft can be divided into major groups of (1) electronics and controls, (2) heating, (3) lighting, and (4) motors. The distribution of electrical loads varies with different airplanes. For example, a large amount of electrical power is required for de-icing. For an airplane that uses engine-bleed air for de-icing, the percent heating load will be relatively low. A comparison of the load division of a few aircraft is given in table I. Detailed load information on a sample SST is given in Appendix B.

TABLE I

COMPARISON OF ELECTRICAL LOAD DIVISION FOR SIX DIFFERENT AIRCRAFT
APPROXIMATE LOAD DISTRIBUTION PERCENTAGE

Load group	Boeing 737	Douglas DC8	SUD-BAC Concorde	Sample SST	Lockheed L-1011	Boeing 747	Approx. average
Electronics and control	13	8	14	11	13	11	12
Heating	42	25	47	11	41	59	37
Lighting	10	12	7	10	10	5	9
Motor	35	55	32	68	36	25	42

The electrical loads of an aircraft can be further classified as essential or nonessential. An example of the division between essential and nonessential loads of an aircraft is shown in table 2. For this particular aircraft the essential and nonessential loads are about equal. A more detailed breakdown of these loads is given in table 3.

Load profile.--An example of the load profile for a large subsonic aircraft is given in table 4 and is plotted in fig. 11.

Power quality requirements.-- Most electrical loads in an airplane do not require good quality input power. Heating loads are insensitive to input waveform and voltage transients and are independent of input frequency. The heat output is proportional to the square of the input voltage; however, the voltage need not be closely regulated.

Many authors have stated that harmonics in the applied voltage (time harmonics) have little effect on electric motor performance. Electric motors normally are not designed with high magnetic saturation. A slight increase in applied voltage does not result in excessive magnetizing current. When the applied voltage decreases, the motor usually draws more current. However, since it takes time for the motor heat to accumulate, electrical motors can usually tolerate relatively large steady-state and transient voltage and frequency excursions.

TABLE 2

SAMPLE AIRCRAFT CONNECTED LOADS

Type of load	Essential		Nonessential		Total (essential and nonessential)	
	kVA	Percent of total connected load	kVA	Percent of total connected load	kVA*	Percent of total connected load
Motor	58.2	28.5	54.9	26.8	113.1	55.1
Heating	6.9	3.4	43.7	21.4	50.6	24.7
Lighting	7.9	3.9	15.9	7.8	23.8	12.7
Electronic	6.8	3.3	0	0	6.8	3.3
Relays, instrument and miscellaneous	2.6	1.3	1.4	0.7	4.0	1.4
TR units	5.8	2.8	0.1	0.1	5.9	2.8
Total*	88.2	43.2	116.0	56.8	204.2	100.0

*Algebraic addition assumed equal power factor on individual loads

Lighting loads can tolerate variable frequency, poor wave form, and large transients. The fluorescent lights can operate on variable frequency as long as the voltage does not vary. The light output of incandescent and fluorescent lights is affected by the applied voltage. However, this affect is not critical.

The avionics equipment utilizes mainly dc power. If ac power is the input to the equipment, it is converted to dc power internally. Only a small fraction of ac power is used directly to supply synchros and resolvers. To supply most avionics loads, 400-Hz ac power is used because it can be transformed easily and provides isolation. Unless some new devices are developed for ac operation, dc power will continue to be the only type needed in future avionics equipment. The present trend indicates that synchros and resolvers will be replaced by dc digital or analog devices. Synchros are disadvantageous for position sensing because they (1) need angular position form of information, (2) do not lend themselves to electronics types of equipment, (3) required precise power with regard to frequency and waveform, (4) have brushes, (5) are heavy, and (6) require more wiring.

Regulated power is not required for some avionics equipment. Equipment that does need regulation is designed for ±1 percent voltage variation and is free from transient excursions. This refined power is obtained by an additional regulator built inside the avionics package. On some occasions, capacitors are also provided to fill-in energy during voltage dips. Digital computers cannot tolerate power interruptions. A short-duration power discontinuity for the automatic landing system while the aircraft is approaching the runway could be disastrous.

TABLE 3

ELECTRICAL UTILIZATION EQUIPMENT OF A LARGE SUBSONIC AIRCRAFT

System	Number of units	Phase	Essential or nonessential	Connected load	
				W	VA
Electronics load					
Air conditioning	7	1	E	69	89
Communication	1	1	NE	48	53
	1	1	E	36	40
	4	3	E	2,000	2,222
Fire protection	6	1	E	20	26
Flight control	3	1	E	211	234
Fuel	2	1	E	154	182
Ice and rain protection	3	1	E	12	15
Navigation	41	1	E	3,002	3,759
Pneumatic	4	1	E	64	80
Engine indication	4	1	E	56	76
Heating load					
Air conditioning	8	1	E	260	260
Galley	6	1	NE	3,360	3,360
	5	1	E	625	660
	1	3	E	125	132
	6	1 (200 V)	NE	21,000	21,000
	4	3 (200 V)	NE	8,200	8,632
Ice and rain protection	27	1	E	4,574	4,574
	6	1 (200 V)	E	4,020	4,674
Instrument	1	1	E	270	270
Flight control	1	1	E	155	155
Fuel	2	1	E	1,600	1,600
Water/waste	6	1	NE	5,280	5,280

TABLE 3.-- Concluded

System	Number of units	Phase	Essential or nonessential	Connected load	
				W	VA
Transformer load (includes lighting)					
Lighting, exterior	10	1	E	5,941	5,941
Lighting, flight compartment	12	1	E	1,278	1,471
Lighting, passenger accommodation	24	1	NE	13,277	13,277
Light, service	3	1	NE	2,051	2,051
Navigation	4	1	E	190	208
Miscellaneous	6	1	E	6	6
T-R	4	3	E	4,295	5,754
	4	1	NE	120	122
Ac motor load					
Air conditioning	14	1	E	1,128	1,475
	1	1	NE	17	21
	1	3	E	1,280	1,600
	4	3	NE	41,270	49,208
Galley	4	3	NE	1,110	1,480
Flight control	1	1	NE	575	676
	3	3	E	8,230	10,038
Fuel	1	1	E	311	518
	12	3	E	10,200	17,004
	1	3	NE	200	286
Hydraulic (emergency use)	6	3	E	22,950	27,000
Lighting control	2	1	E	368	368
Pneumatic	4	1	E	184	232
Water/waste	7	3	NE	2,594	3,244
Relay indicator and miscellaneous					
Relay	41	1	E	734	826
	13	1	NE	192	217
Indicator	31	1	E	120	193
Miscellaneous	38	1	E	1,408	1,615
	14	1	NE	1,160	1,200

TABLE 4

ELECTRICAL LOAD PROFILE IN KVA

System	Connected load	Loading		Start	Taxi	Takeoff and climb		Cruise		Descent and landing	
Lighting	23.8	15.6	13.1	14.8	16.0	19.5	18.4	15.8	15.5	20.0	18.4
Navigation and flight control	17.7	2.2	2.1	2.4	8.1	9.7	5.7	4.3	4.3	12.8	13.0
Transformer rectifier	5.8	3.2	3.2	4.0	2.6	2.9	2.8	2.0	2.0	3.3	3.0
Ice and rain protection	10.0	2.6	1.5	2.5	3.9	3.5	5.6	4.4	2.7	5.5	5.6
Fuel	19.8	.3	.3	11.7	15.9	15.9	7.7	10.7	10.3	6.0	6.0
Instrument	1.9	1.5	.8	.5	.5	.5	.5	.5	.5	.5	.5
Galleys	36.2	4.6	4.6	3.1	4.6	4.6	11.4	19.6	19.6	4.0	4.0
Lavatory	8.5	2.3	1.8	3.2	3.2	3.2	3.2	3.8	3.8	3.2	3.2
Air conditioning	53.5	52.2	50.9	52.0	52.2	52.2	35.1	20.0	20.0	52.3	30.9
Hydraulic (for emergency)	27.0										
Total*	204.2	84.5	78.3	94.2	107.0	112.0	90.4	81.1	78.7	107.6	84.6

*Algebraic addition assuming equal power factor on individual loads

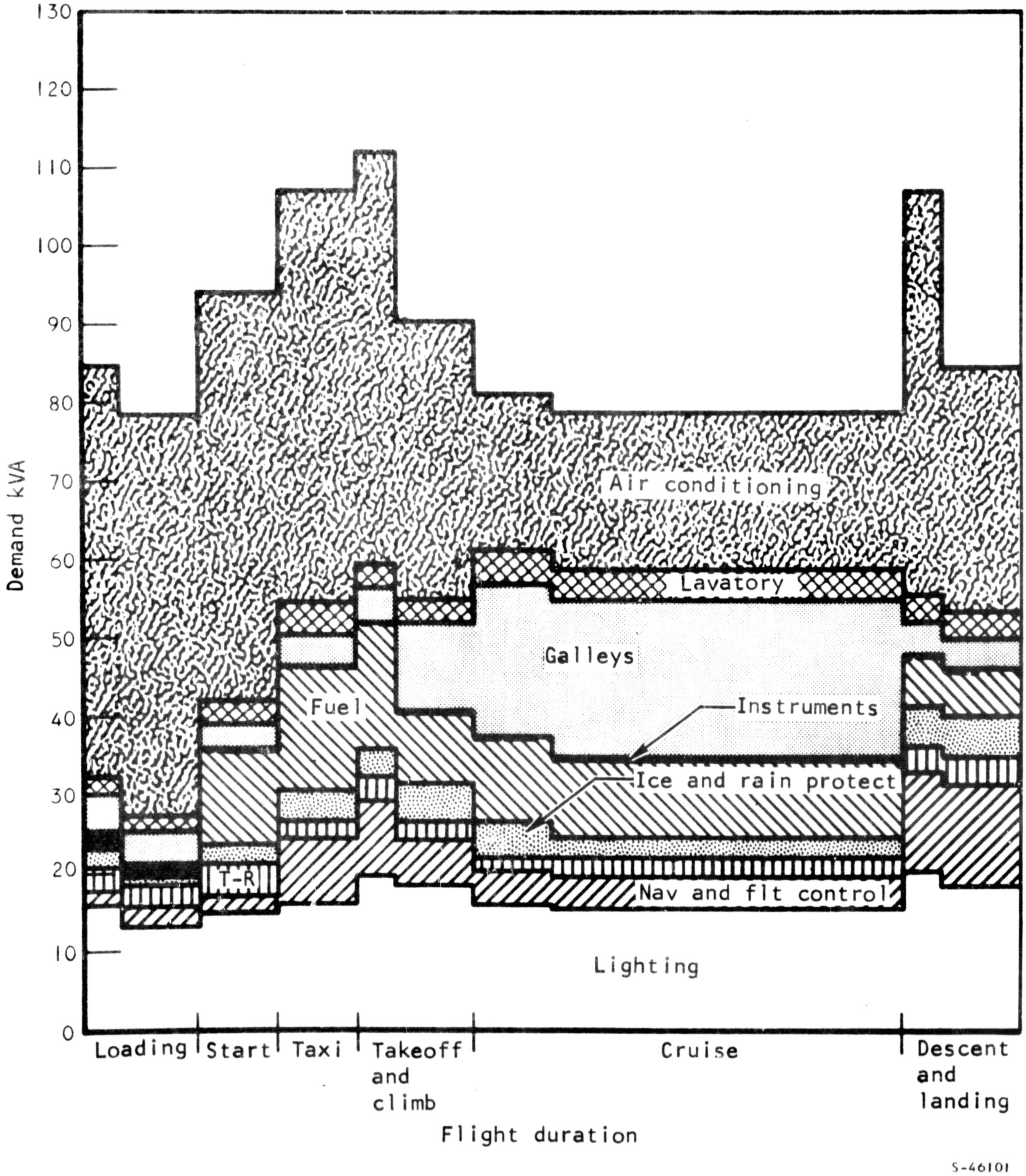
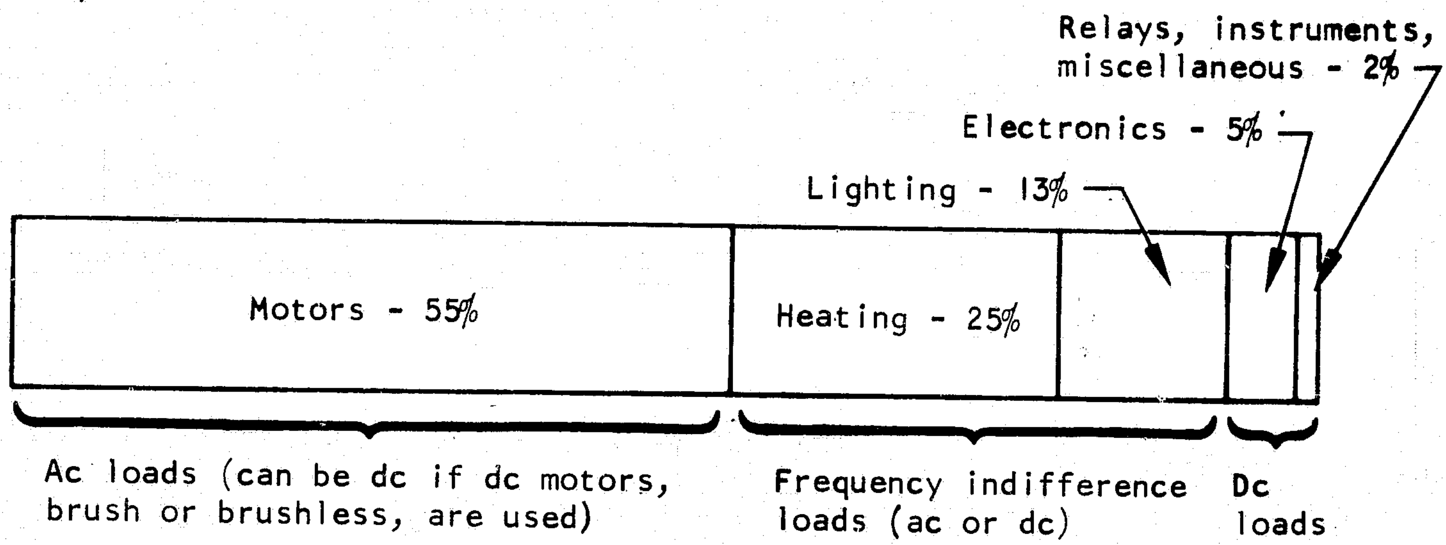


Figure 11. Large Subsonic Aircraft Electrical Power Demand

One avionics manufacturer revealed that additional regulation inside the electronics package is necessary mainly because of the poor ground power characteristics. Some avionics devices could be unjustifiably penalized because of the poor ground checkout equipment. A detailed investigation and study would more clearly define the problem areas and provide possible solutions.

The dominant characteristic of electrical loads on a large subsonic aircraft can be summarized by the following diagram:



In other words, about 7 percent of the loads is consumed in the form of dc power, while the other 93 percent can be consumed as either ac or dc.

Comparison of Electric, Hydraulic, Pneumatic, and Mechanical Drive Systems

Many aircraft loads are presently supplied by either hydraulic or pneumatic power sources. These loads may be supplied either directly (electric actuators) or indirectly (electric driven hydraulic pump) by the electric power system. Consequently, a number of possible tradeoffs for the provision of electric, hydraulic, pneumatic, or mechanical power to the aircraft loads can be identified. Generally hydraulic drive is better for loads that require a large amount of power for a short duration, while electric drive is better for long duration loads. The various types of drive systems are discussed in the following paragraphs.

Mechanical systems.-- Power distribution to the output members of an actuation system is provided through a rotating shaft by high-speed, low-torque mechanical drives. At the controlled surface, the power is converted to a high-torque, but relatively low-speed output. The power conversion is accomplished through a coupler linking the power source to the mechanical system, a mechanical servo, transmission components, and an actuator. Component such as gearboxes, bearings, shafts, and couplings needed for power transmissior and pushrods, bell cranks, and other relatively heavy linkages required for the flight control mechanism result in a bulky assembly. For example, the F-111 pitch/roll system has 114 bearing points that are required to support the linkages. The B-70 flight control system is even more complex than the F-111, and the helicopter systems are also very complex.

The use of mechanical actuation and flight control systems is becoming more limited as current and future aircraft increase in size, power requirements, and speed. As power transmission lines increase in length and power requirements become higher, the use of mechanical drives becomes less attractive. The mechanical servo component has a poor response time and relatively low accuracy. Although the claim for high reliability is often made for mechanical actuation systems and controls, it has not been demonstrated, especially under the duty cycle and environmental conditions of primary flight control system requirements.

Isolated examples of recent development programs indicate an interest in developing improved mechanical systems and components. Lycoming has provided a 600°F (320°C) constant speed drive for the North American Rockwell, USAF HOTELEC program. Subsequently, this mechanical principle was applied to an integrated hydraulic and electric power and starting system for the Republic "Swallow" drone and the Douglas A4D aircraft.

The United Shoe Machinery Corporation has developed a harmonic gear drive mechanical actuator which has been licensed to Bendix Eclipse-Pioneer Division for application to aircraft actuation systems, specifically the Bell X-22A aircraft. Curtiss-Wright, Propeller Division, (ref. 1) has developed an epicyclic gear train mechanical actuator which has been used for the XB-70 wing fold actuator.

It is doubtful that mechanical systems will ever be developed to the degree required for effective competition with other forms of power transmission and controls. Mechanical systems have advantages and disadvantages, however, which must be considered if an optimum system is to be developed for future aircraft. Advantages include high stiffness, no fluids (except lubrication), no sealing problems, low weight (power transmission), and direct coupling. Disadvantages include complex installation (especially controls), heavy weight (controls), poor response, poor accuracy, inherent wear, heat generation, low reliability (servo), increased volume requirement, and subject to thermal expansion and flexing.

Utilization of mechanical systems on future aircraft will probably be limited to short distance power transmission from the prime mover. Mechanical flight controls, actuation systems, and long distance power transmission does not appear attractive.

<u>Hydraulic and pneumatic systems</u>.-- Systems that use hydraulic or pneumatic drives as a primary source of power include flight controls, landing gear, thrust reversers, and engine starting. The primary flight control surfaces (ailerons, spoilers, rudder, and elevators) are ordinarily activated by hydraulic cylinders. Secondary flight control surfaces (leading edge slat and trailing edge flaps) are usually actuated by hydraulic motors with screwjack mechanisms. Pneumatic power is also used for flight control purposes. Hydraulic power is used for retracting and extending the landing gear, steering the nose landing gear, and operating the brakes. Mechanical or electrical actuator backup provisions are ordinarily incorporated into the system for emergency lowering of the landing gear. Redundancy is provided in the hydraulic system and brakes to ensure availability of braking at a high probability. Pneumatic

power is not used for landing gear operations. The actuators used to position thrust reversal devices are operated either hydraulically or pneumatically. Small engines (helicopter power plants or gas turbine APU's) are often provided with hydraulic starters. Large jet engines usually are equipped with turbine starters which use pneumatic power provided by the APU for engine starting. To facilitate comparison, the hydraulic systems in existing airplanes as well as some parametric information of current hydraulic equipment are given in Appendix F.

Interest in pneumatics has arisen primarily in an effort to alleviate the problems presently associated with hydraulics, namely high temperature. Although the basic components of a pneumatic system are similar to those of a hydraulic system, the characteristics are significantly different for three primary reasons (ref. 2).

(1) The flow through an orifice in a pneumatic valve becomes sonic with large pressure differentials.

(2) The compressibility and lower viscosity of the gas result in higher discharge coefficients and leakages.

(3) The compressibility of the gas requires servovalve design techniques that give good bandwidth and stiffness.

One pneumatic component that is substantially different from the hydraulic counterpart is the power source. Pneumatic power is realistically obtained by a number of practical means, some of which are listed below.

(1) Compressed gas storage

(2) Power plant bleed air

(3) Gas generator

Solid propellant
Liquid bipropellant
Liquid monopropellant

(4) Air compressors

(5) Ducted ram air

(6) Cryogenic storage bottles

(7) Chemical decomposition

Selection of the gas power source requires knowledge of the application from both a quantity and time standpoint. In addition, the system pressure level is an important consideration. Air compressors (possibly ram-air fed) will probably prove to be the most practical primary power source on supersonic aircraft. Although bleed air is readily available from the power plant compressors, its use has been estimated to cost three times the range penalty in required pounds of extra fuel when compared to costs occurred from using the shaft power drive of a compressor.

The same component is used in a system for power control as is used in a hydraulic system. Pneumatic servovalves are available in several component types including flapper, poppet, jet pipe, sliding plate, and spool. The poppet and sliding plate designs are the most desirable since both the flapper and jet pipe have high quiescent flow and are suitable to only relatively low-power levels, while the spool design tolerances are sensitive to contaminants.

Another technique worthy of additional study is the fluidic amplifier. Adaptation of this concept to provide the servo function would greatly reduce the complexity of the servo and eliminate many of the moving parts. Many companies have actively studied fluidics for some time and the applications appear to be widespread.

The pneumatic power output device is usually a rotary one, but can be a linear actuator similar to the hydraulic actuator. A rotary device is used to enable incorporation of a gearset which adds stiffness to the actuator. The rotary devices consist of rotary piston motors, cam-piston motors, gear motors, rack and pinion motors, nutating-disc motors, and rotary vane motors. Considering the requirements of future aircraft, the nutating-disc motor and the cam-piston motor designs offer the greatest potential for continued development.

Pneumatic systems have received development interest primarily because they are more readily adaptable to high temperature operation and to a wide range of environments. Without extensive system design complexity, however, pneumatic systems lack stiffness and response due to compressibility of the gas. If, by design complexity, system stiffness is achieved, then system stability becomes a problem due to stiction. Applications of pneumatic systems, therefore, have been limited to functions of a secondary power nature.

Again, as in mechanical systems, pneumatics have received isolated development interest. The most noteworthy example is the all-pneumatic flight control and actuation system development by Bendix (ref. 3) and General Dynamics. The hardware design problems encountered in this program make it uncertain whether an all-pneumatic flight control system could be developed to meet the requirements of the supersonic transport. Pneumatic actuation systems will undoubtedly find application where dynamic performance is less critical or extreme environments are obtained. This type of application is exemplified by the actuators used to reverse the thrust of the propulsion engines.

Electrical systems.--Electric systems are described in detail in other sections of this report. Conventionally, the primary source of electrical power is usually obtained from generators CSD-driven from the accessory pad of the aircraft main power plant or from an accessory power plant or gearbox. This transmission system is a combination of hydraulic and electric power.

Electrical actuators are suitable for applications where moderate loads and moderate response are required with low stiffness and low torque-to-inertia ratios. These disadvantages and the weight penalty, however, may be offset by requirements for reduced envelope, reduced maintenance, and lower cost factors.

System comparison.--In a study by McJones and Gangnath (ref. 4), comparisons were made of various systems weights and efficiencies as functions of the power delivered. The power conversion unit weights are presented in fig. 12; the efficiencies are presented in fig. 13. These data were determined on the basis of a group of representative engine specifications including a design at Mach 3 operation. From the power conversion standpoint, the hydraulic system has both a higher efficiency and a lower specific weight than either the electrical or the pneumatic system.

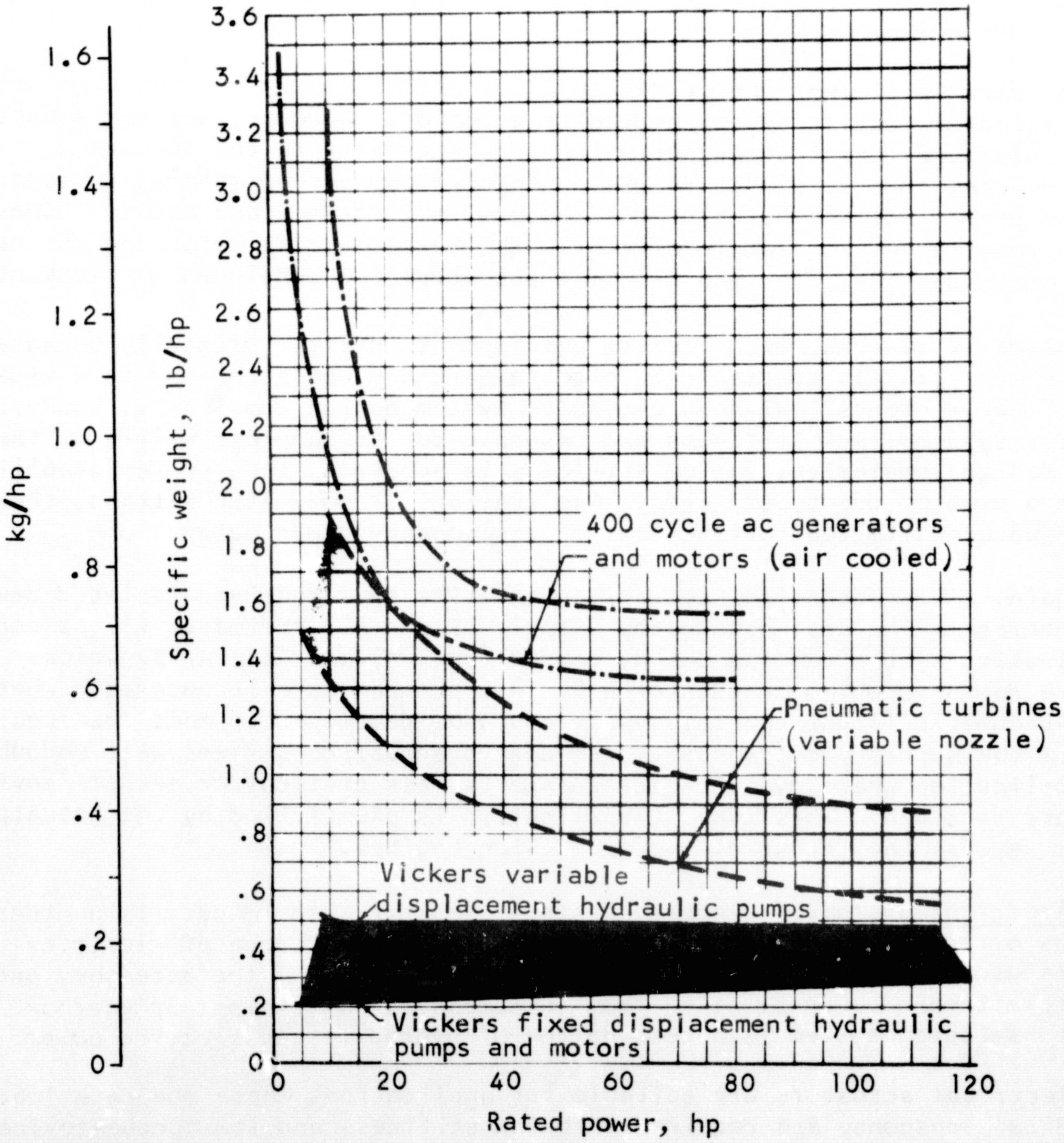


Figure 12. Power Conversion Unit Weights (ref. 4)

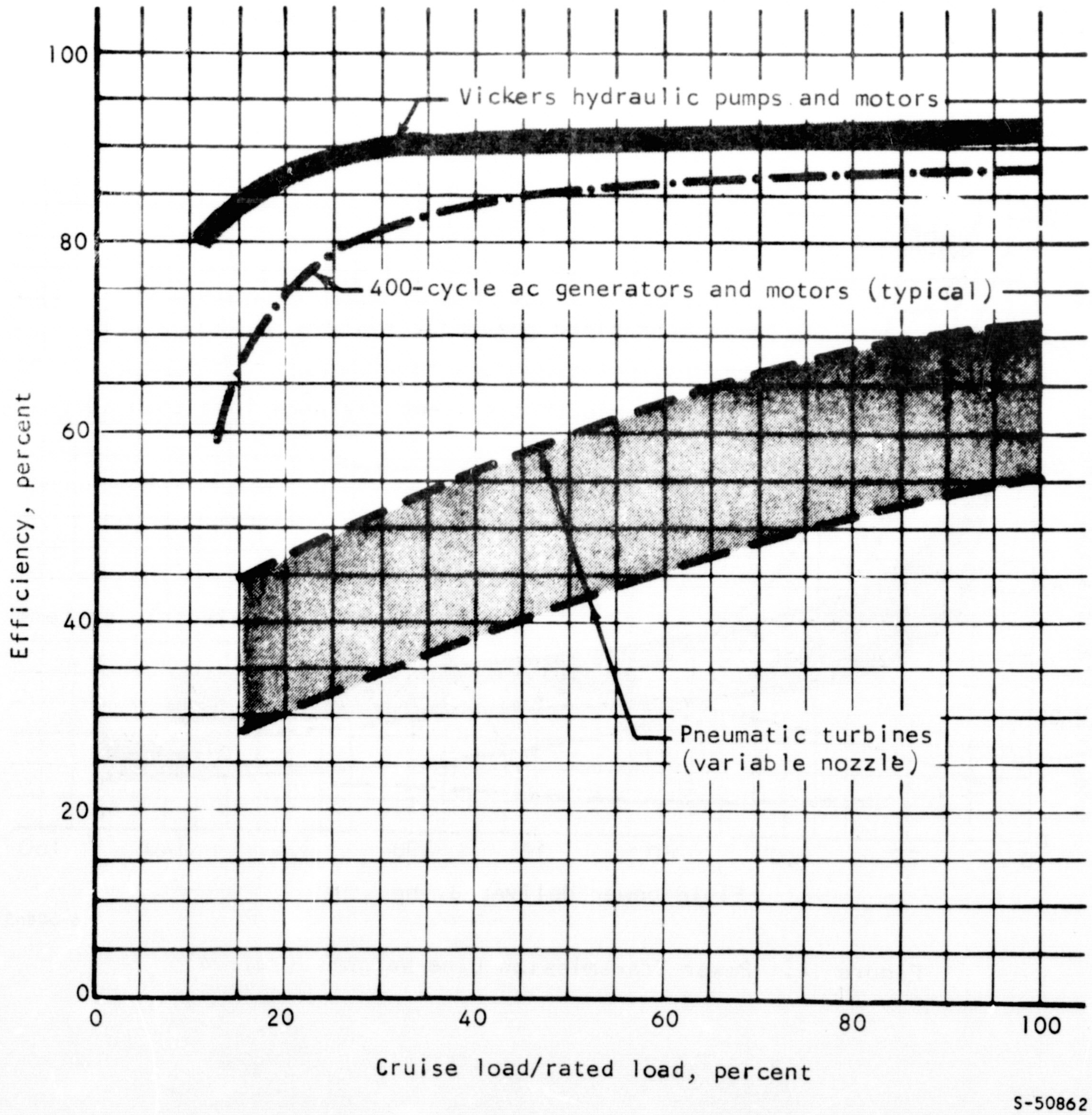


Figure 13. Power Conversion Unit Efficiencies (ref. 4)

Hydraulic system power transmission is reasonably competitive as shown in fig. 14, but the efficiency of power transmission drops off rapidly with increasing distance, fig. 15.

A qualitative summary of the comparison of the four systems, i.e., mechanical, hydraulic, pneumatic, and electrical, is presented in table 5.

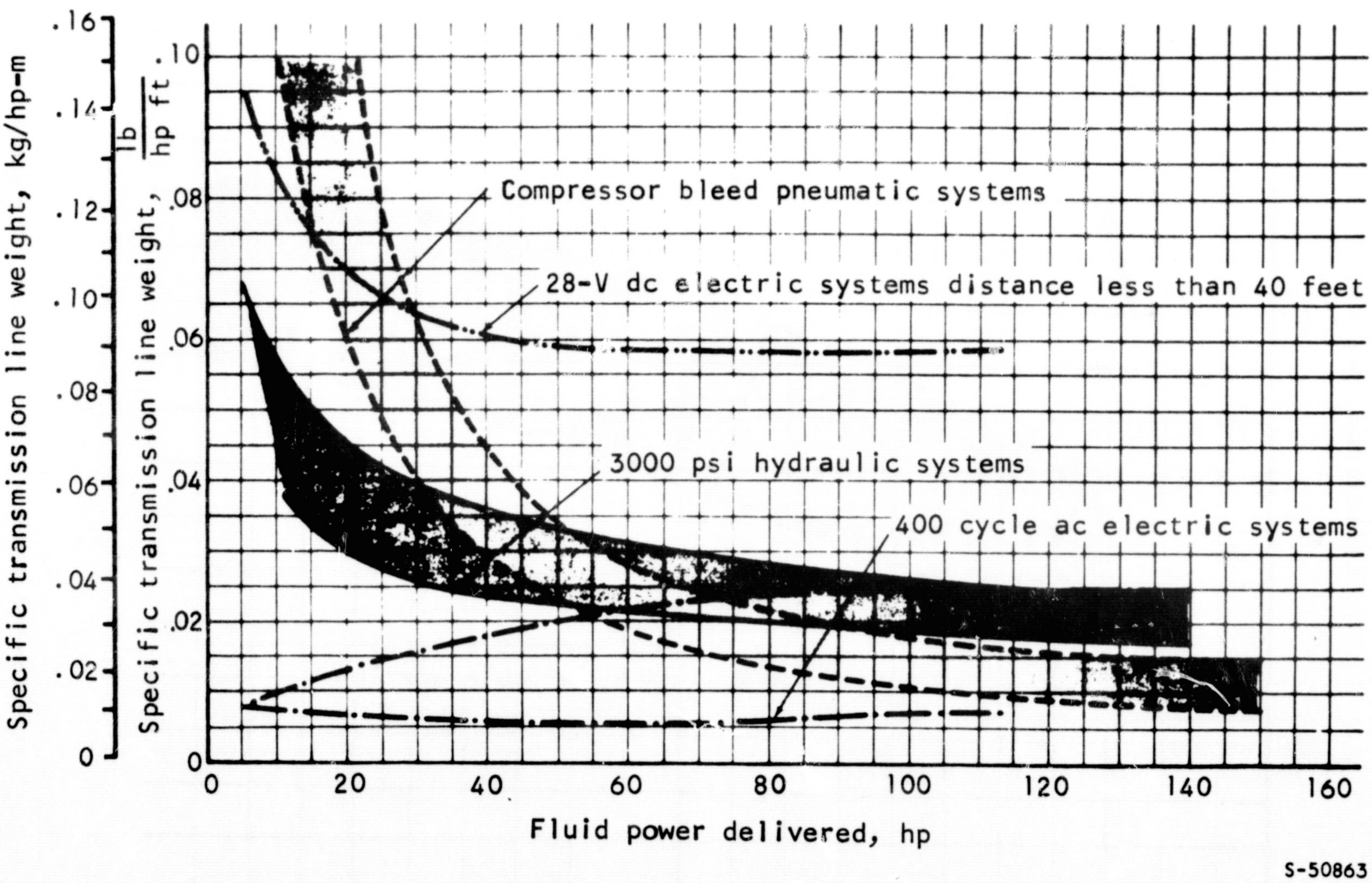


Figure 14. Power Transmission Line Weights (ref. 4)

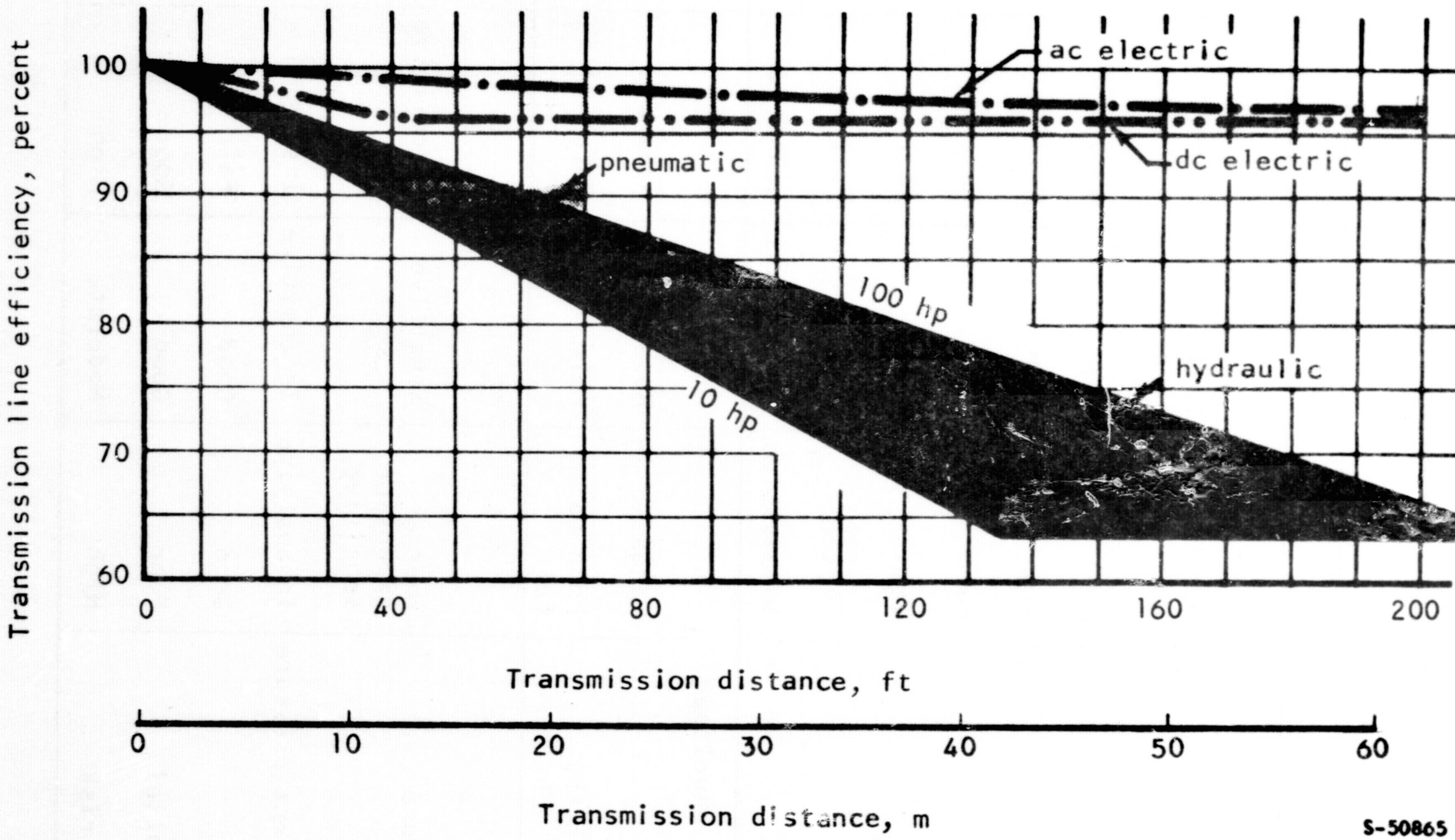


Figure 15. Transmission Line Efficiencies (ref. 5)

TABLE 5

SUMMARY OF SYSTEMS CAPABILITY

Factors	Mechanical	Hydraulic	Pneumatic	Electrical
Load				
Magnitude	Moderate-unassisted	Full range	Light to moderate	Light to moderate
Rate	High	High	Moderate	Moderate
Response	High	High	Fair	Fair
Accuracy	Fair	Good	Fair	Fair
Stiffness	High	High	Poor	Poor
Stability	Good	Good	Fair	Fair
Inertia	Insensitive	Insensitive	Sensitive	Sensitive
Installation envelope				
Volume	Large	Moderate	Moderate	Minimum
Accessability	Poor to fair	Fair	Fair	Good
Environment				
Temperature	Thermal expansion sensitive	Sensitive	Insensitive	Sensitive
Low ambient pressure	Insensitive	Insensitive	Insensitive	Sensitive
Reliability	Good	Good	Fair	Fair
Growth potential	Poor	Good	Good	Good
Development risk	High	Moderate	High	Moderate

2

Component Parametric Data

Induction motors.--A large number of ac induction motors are used in existing aircraft because they are simple, rugged, lightweight, and require almost no maintenance. These motors drive fuel boost pumps, hydraulic pumps, blowers, freon and air compressors, and actuators. Current aircraft ac motors are run on a 400-Hz, 115/200-Vac source. In fraction and subfraction horsepower ranges, the motors are usually a single phase design. In integral horsepower ranges, they are usually 3-phase machines.

Induction motors are very reliable; their mean time between failures (MTBF's) are 5000 hr or more. Efficiencies and power factors vary with the type of design and the kind of application. Generally, the efficiencies range from about 50 percent for the 1/20 hp size to about 85 percent for the 15 hp size. The power factors are usually in the same order of magnitude as the efficiencies.

The weight parametric curves of induction motors are obtained from the motor weight information in trade catalogs of General Electric, Westinghouse, Western Gear, etc., in addition to the AiResearch motor data. The average weight per horsepower versus horsepower for 400 Hz aircraft induction motors with various speeds is shown in fig. 16. The average weight per horsepower versus speed with horsepower as parameter was also obtained and plotted in fig. 17. Since the slopes of curves in fig. 17 are relatively constant, they are combined as one curve for approximation purposes as shown in fig. 18. The approximate efficiency and power factor of these motors are shown in fig. 19. Figs. 18 and 19 are for 3 phase induction motors compatible with present aircraft motor requirements. They represent current production motors with silicon-steel laminations and class H insulation. The data represents typical or average cases; the weight of a motor with given horsepower and speed can vary considerably depending on its efficiency, power factor, overload capability, starting performance, ambient conditions, method of cooling, and other factors.

Dc motors.--Many brush type dc motors are in use in existing aircraft. Primarily, they are installed in small aircraft and in systems that relay on battery power such as gas turbine starters. Dc motors have better starting characteristics than ac motors, although they are heavier and require maintenance on brushes.

The average pound per horsepower is plotted in fig. 20 from a large number of aircraft 28-Vdc motors described in trade catalogs.

Gearbox weight estimate.--Although the weight of the electric motor can be reduced by increasing the speed, the high operating speed may not be suitable to drive the load directly; a gearbox may be required. In optimizing the operating speed for a certain drive, the total weight of the package (including motor, gearbox, load, and supporting structure) should be considered. If low output speeds are required, weight savings may be accomplished by using reduction gears with high-speed motors rather than using low speed motors directly. The life of a reduction gear is almost an order of magnitude longer than that of an electric motor. The reduction in reliability by adding a gearhead to the motor is relatively small.

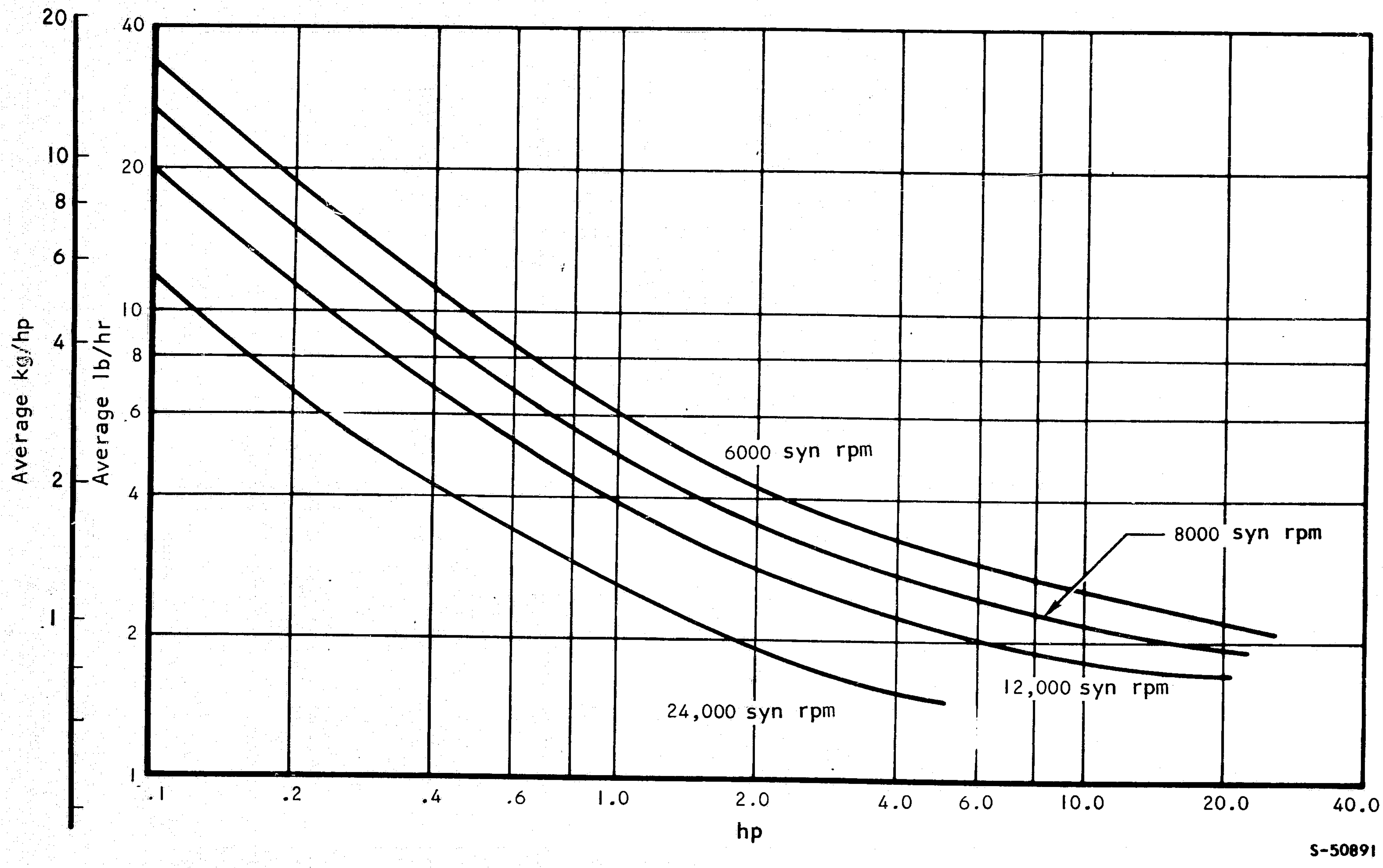


Figure 16. Average lb/hp versus hp, Aircraft AC Induction Motors, 400 Hz, 115 v/phase

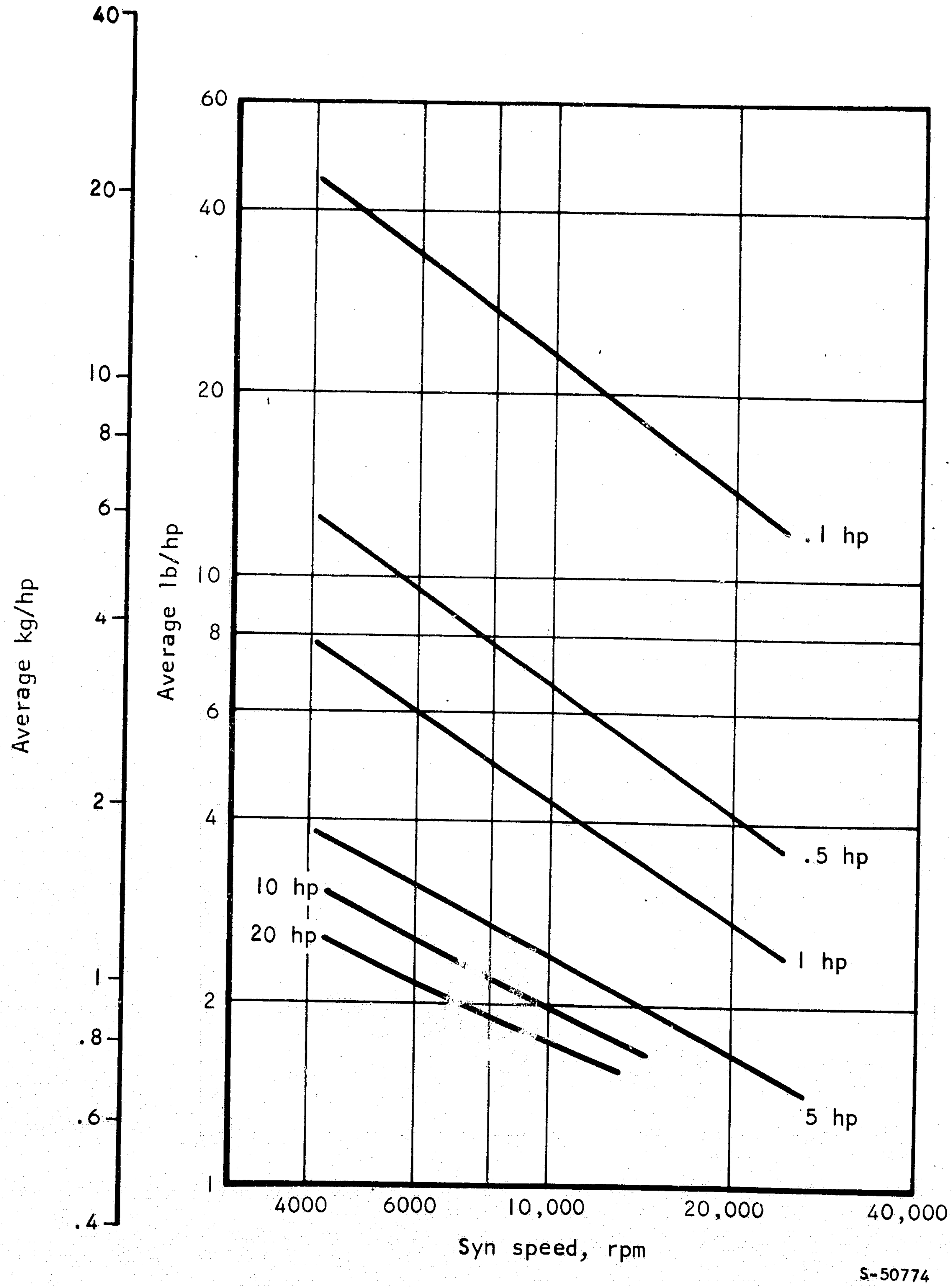


Figure 17. Average lb/hp versus Synchronous rpm, Aircraft AC Induction Motors, 400 Hz, 115 v/phase, Conventional Silicon Steel Laminations

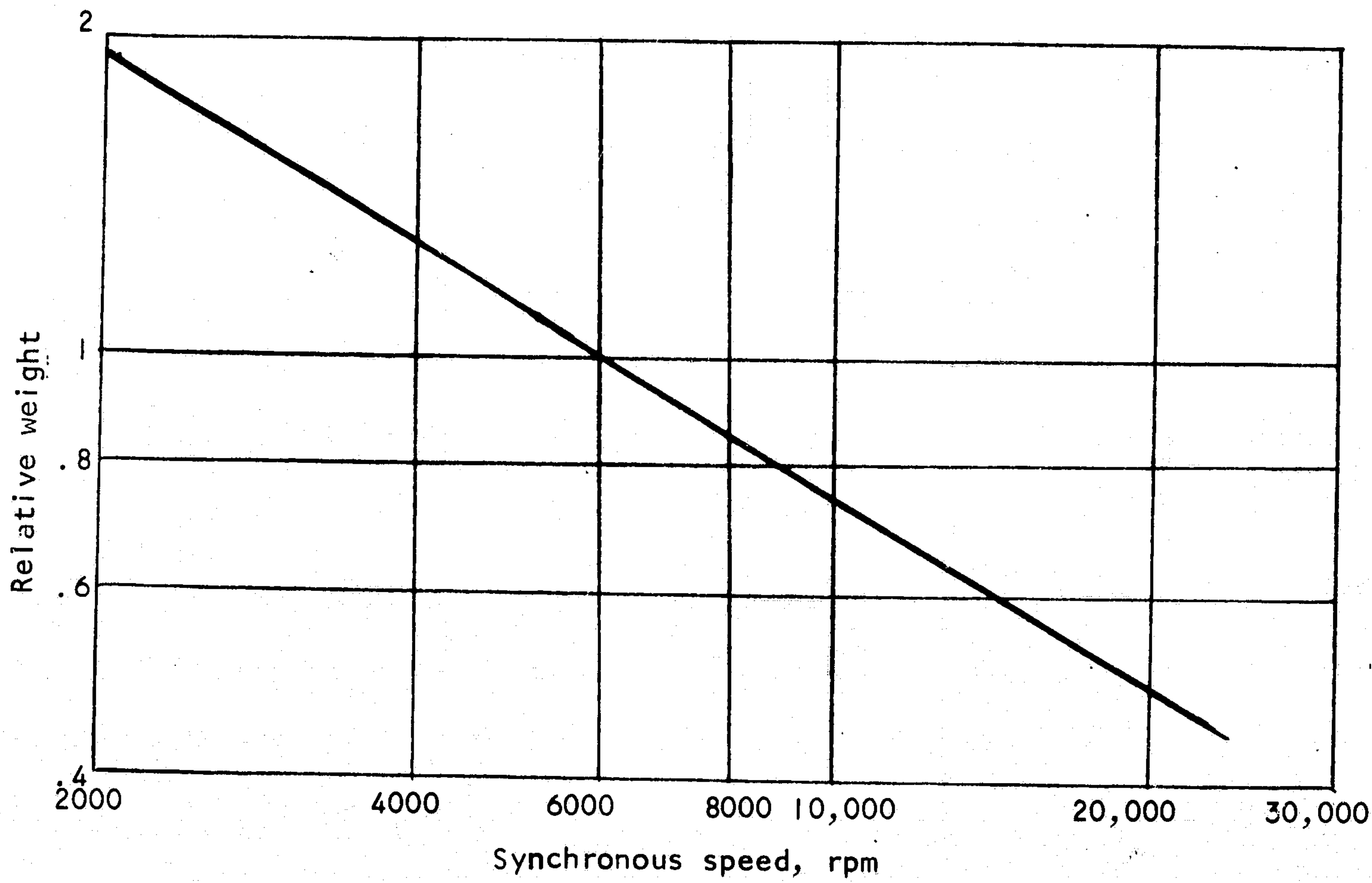


Figure 18. Average Relative Weight of Induction Motor vs Speed, 400 Hz, 3 Phase Motors, 0.1 to 25 hp, Conventional Materials

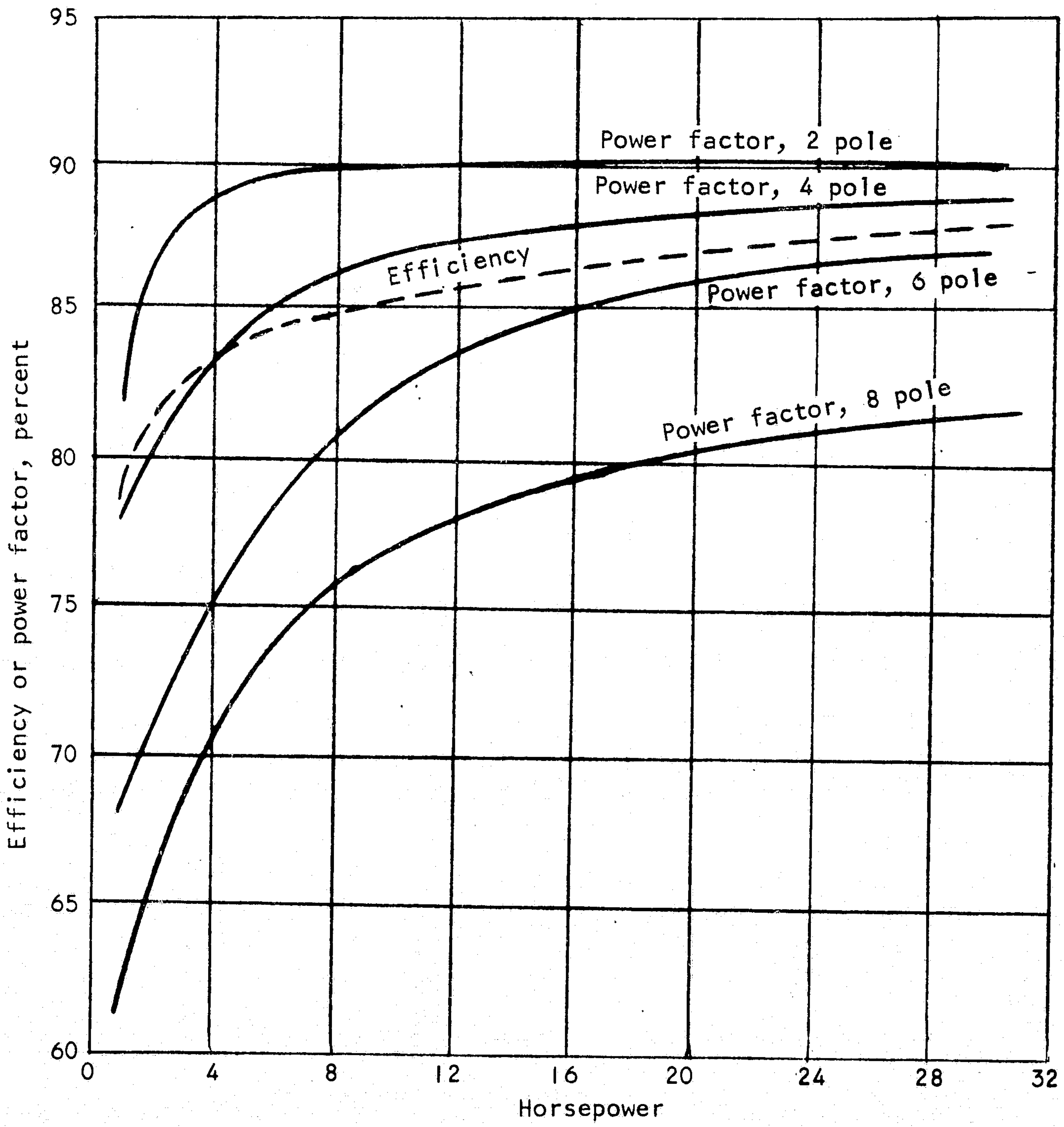


Figure 19. Approximate Efficiency and Power Factor, 400 Hz, 3 Phase Aircraft Induction Motors, Conventional Materials

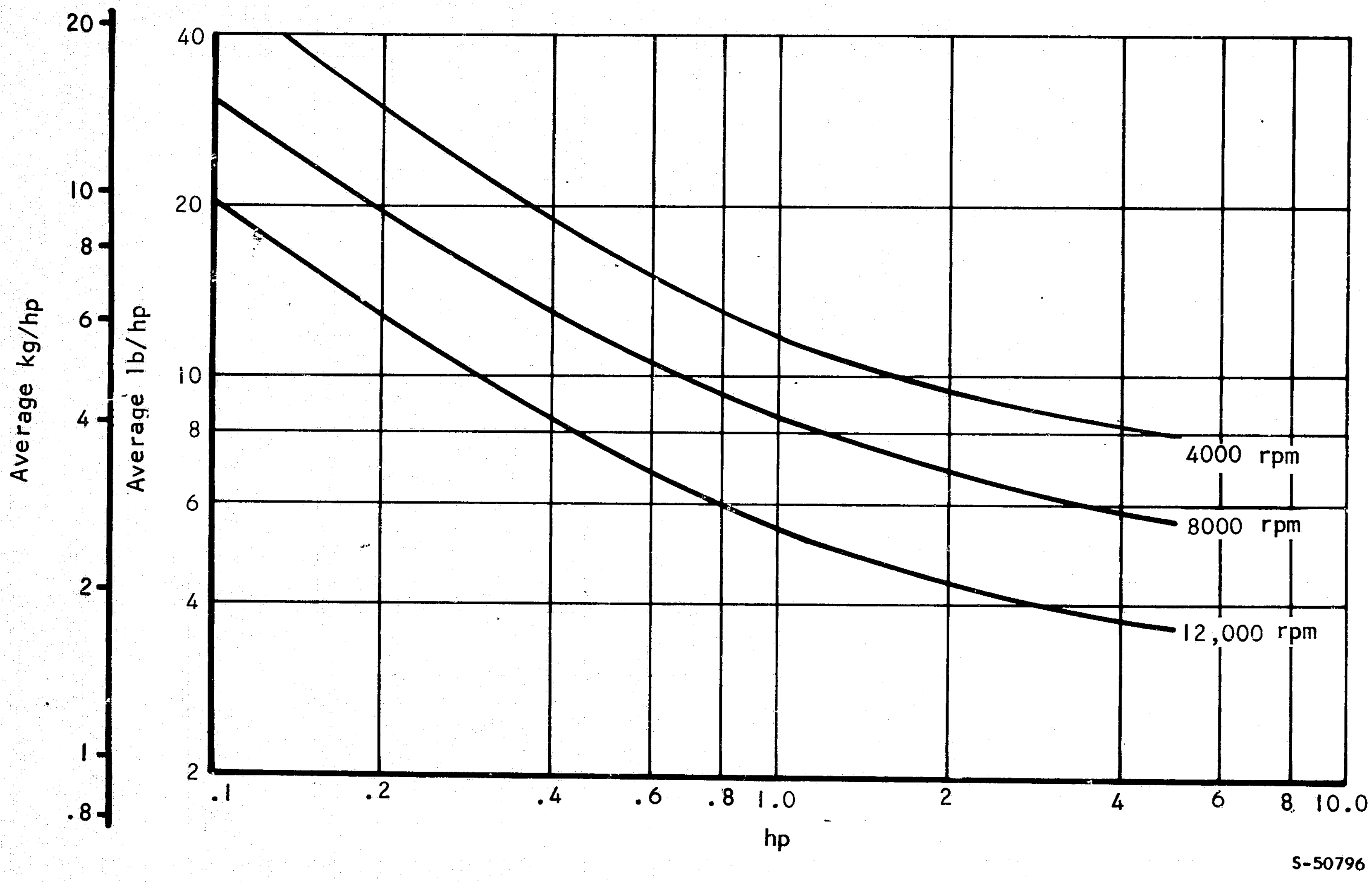


Figure 20. Average lb/hp versus hp, 28 v Aircraft DC Motors

A method of estimating the gearbox weight developed by D. W. Dudley for single stage gearing used in current production is given below (ref. 6).

According to Dudley, gear weight is a function of the Q-factor defined as:

$$Q = \frac{hp}{R}\,\frac{(N + 1)^3}{N}$$

where hp = horsepower transmitted

R = pinion rpm

N = gear ratio

Since the pinion rpm is N times the output rpm,

$$Q = \frac{hp}{R_o}\,\frac{(N + 1)^3}{N^2}$$

where R_o = gearbox output rpm

Gearset weight is also a function of the K-factor which is an index for measuring the intensity of tooth loads. A curve relating the Q-factor and the gearset weight for a K-factor of 500 is extrapolated from ref. 6 and is shown in fig. 21. This curve can be represented by

$$W_{gear} = 192\ Q^{.807}$$

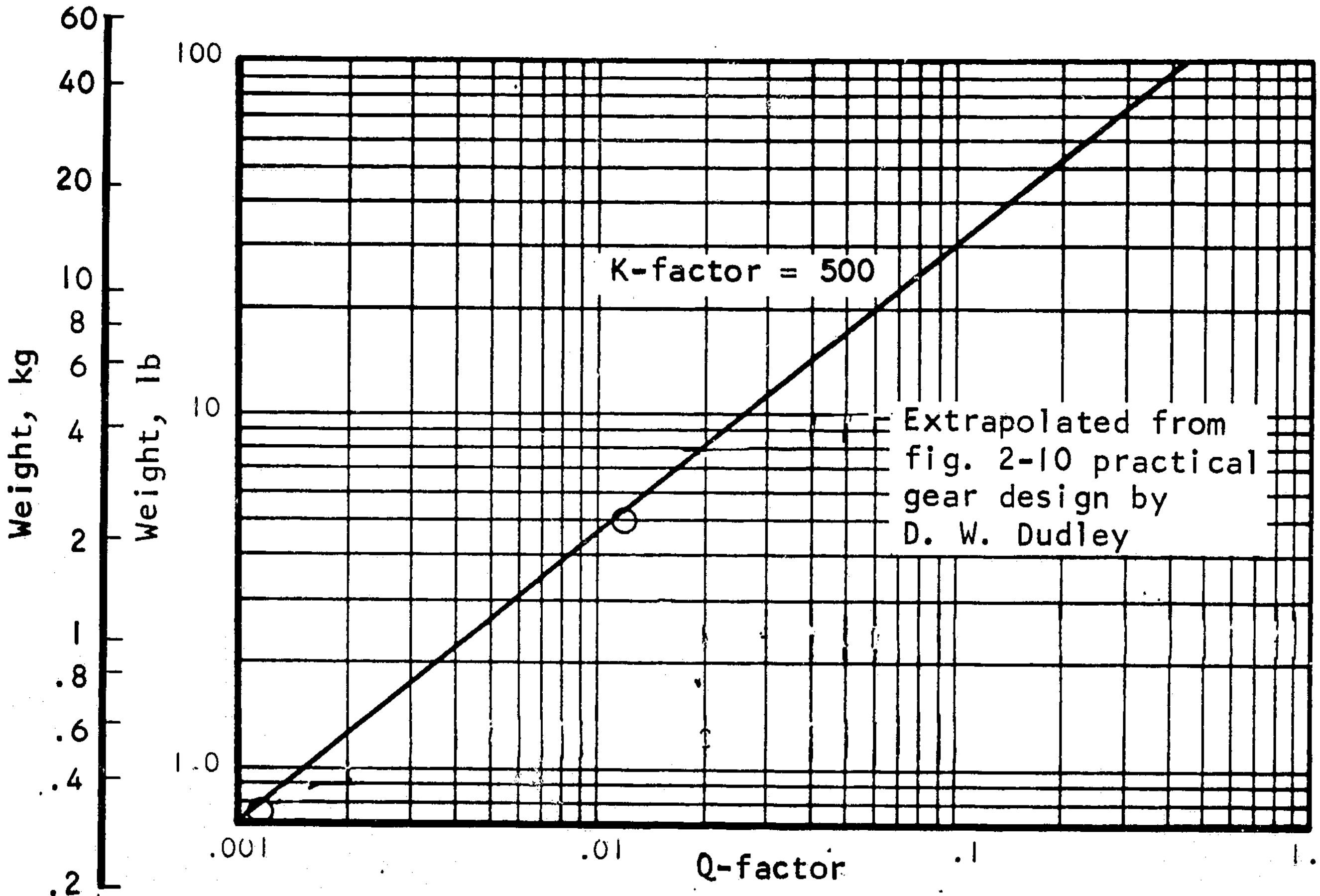
$$= 192 \left(\frac{hp}{R_o}\right)^{.807} \left[\frac{(N + 1)^3}{N^2}\right]^{.807}$$

Transformers.--The weight of an aircraft transformer of a given kVA rating will vary considerably as a result of design requirements such as temperature rise, efficiency, regulation, method of cooling, etc. The approximate relative specific weights vs kVA rating are shown in fig. 22. A typical 5-kVA aircraft transformer weighs approximately 3 lb/kVA (1.3 kg/kVA) and an efficiency of approximately 97 percent. Transformers have high reliability and long life in comparison with other aircraft electrical components.

The relationships of weight vs kVA, efficiency and input voltage wave form for transformers are shown in the following paragraphs.

Specific weight vs kVA: A common empirical formula used by transformer designer is that

$$\text{Transformer weight} \propto (\text{kVA rating})^{3/4} \qquad (1)$$



$$Q = \frac{\text{hp}}{\text{pinion rpm}} \quad \frac{(\text{speed ratio} + 1)^3}{\text{speed ratio}}$$

Figure 21. Gearbox Weight vs Q-Factor (ref. 6)

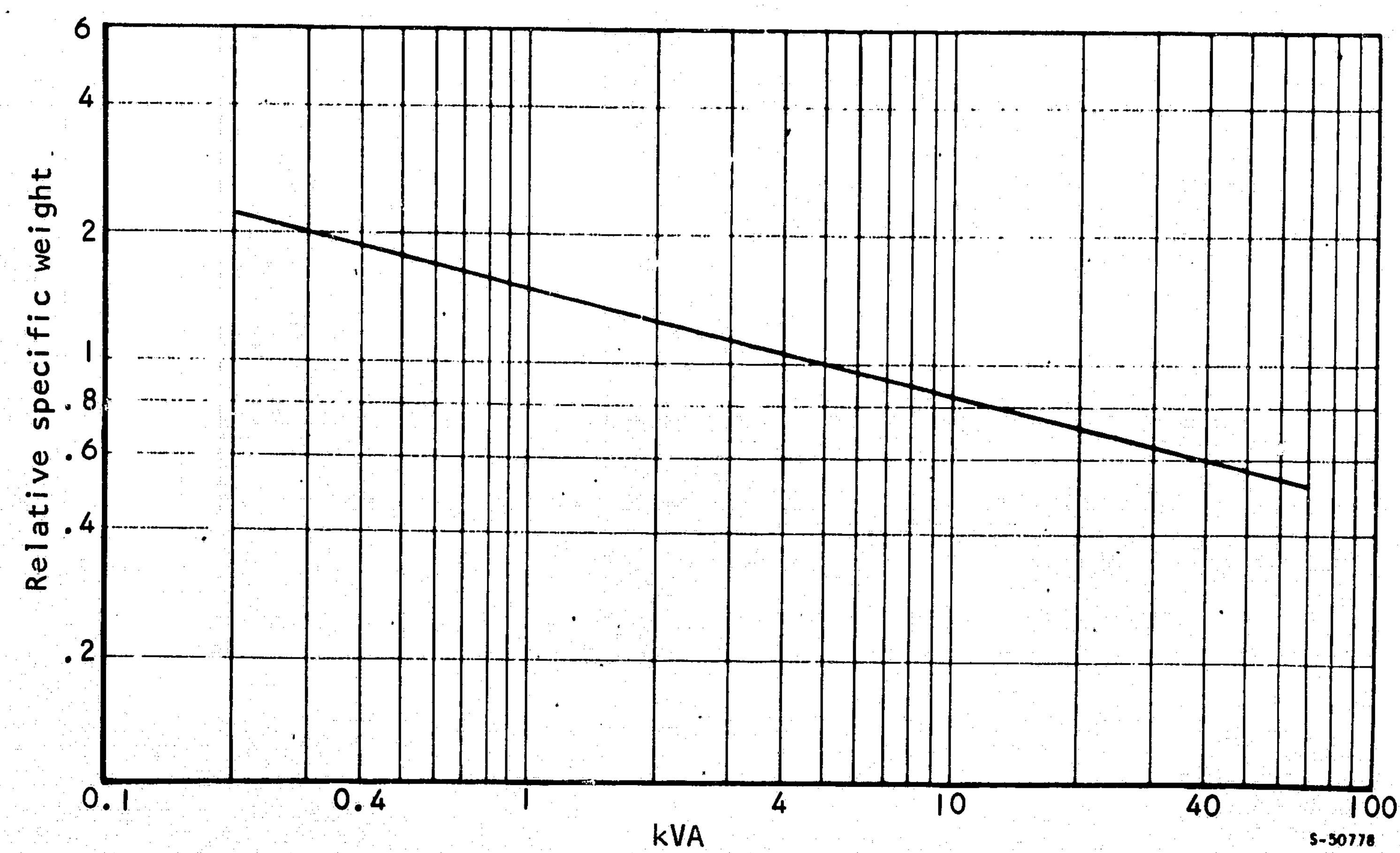


Figure 22. Transformer Weight/Rating Relation

Dividing the formula by kVA,

$$\text{Specific weight (lb/kVA or kg/kVA)} = (\text{kVA})^{-1/4}$$

This shows that the specific weight of the transformer decreases as the kVA rating increases. The above relation is shown in fig. 22.

Weight vs efficiency: An empirical formula used by transformer designers relating weight and efficiency for a given kVA rating is:

$$\text{Weight} \propto \frac{1}{(1-\eta)^{2/3}} \tag{2}$$

where η is the efficiency. This relationship can be verified by transformer design formulas. In ref. 7, eq. (5.9), p. 110,

$$\text{Power} \propto (\text{frequency})^{0.76} (\text{temperature rise})^{0.63} \text{ for a given transformer} \tag{3}$$

At constant frequency:

$$\text{Power} \propto (\text{temperature rise})^{0.63} \tag{4}$$

Combining relations (1) and (4):

$$\text{Power} \propto (\text{weight})^{4/3} (\text{temperature rise})^{.63} \tag{5}$$

However, temperature rise can be expressed as:

$$\text{temperature rise} \propto \frac{\text{losses}}{\text{dissipating surfaces}} \tag{6}$$

and $$\text{losses} = (\text{power})(1-\eta) \tag{7}$$

where η = efficiency

The dissipating surface is approximately related to the size of the transformer as

$$\begin{aligned} \text{Surface} &\propto (\text{volume})^{2/3} \\ &\propto (\text{weight})^{2/3} \end{aligned} \tag{8}$$

Substituting relations (7) and (8) into (6):

$$\text{Temperature rise} \propto \frac{(\text{power})(1-\eta)}{(\text{weight})^{2/3}} \tag{9}$$

Substituting relations (9) into (5):

$$\text{Power} \propto (\text{weight})^{4/3} \frac{(\text{power})^{.63} (1-\eta)^{.63}}{(\text{weight})^{2/3 \times .63}}$$

If power rating is a constant:

$$(\text{weight})^{(1.33-.42)} (1-\eta)^{.63} = \text{constant}$$

In other words:

$$(\text{weight})^{.91} \propto \frac{1}{(1-\eta)^{.63}}$$

or $$\text{Weight} \propto \frac{1}{(1-\eta)^{.69}} \simeq \frac{1}{(1-\eta)^{2/3}}$$

The above relationship between weight and efficiency is given in fig. 23.

Weight vs waveform: For a given value of kVA at a fundamental frequency, the weight of a transformer will be approximately 10 percent lower with the stepped voltage waveform than with the sinusoidal waveform.

Compare the voltage forms applied to transformers shown in fig. 24. Let the amplitude of the sinusoidal wave be 1.0. For the same rms value, the amplitude of the 120-deg stepped wave will be $\sqrt{3}/2$. The peak magnetic fluxes in the transformer cores are proportional to the Volt-sec or areas under the voltage waves. Thus:

$$\text{Volt-sec of sinusoidal wave} \propto 2$$

$$\text{Volt-sec of 120-deg stepped wave} \propto \pi/\sqrt{3}$$

The ratio of the magnetic fluxes is therefore

$$\begin{array}{l}\text{Ratio of peak fluxes}\\ \text{in transformer cores}\end{array} \left(\frac{\text{Sinusoidal}}{\text{Stepped}}\right) = \frac{2\sqrt{3}}{\pi} = 1.11$$

In other words, the area of the transformer core for a 120-deg stepped wave can be 11 percent smaller than that for a corresponding sinusoidal wave. If the core cross-sectional area is smaller, the length of mean turn of the copper winding is also smaller. Design experience has indicated that the transformer for a 120-deg stepped voltage is about 10 percent lighter than that for a sinusoidal voltage of the same rms value.

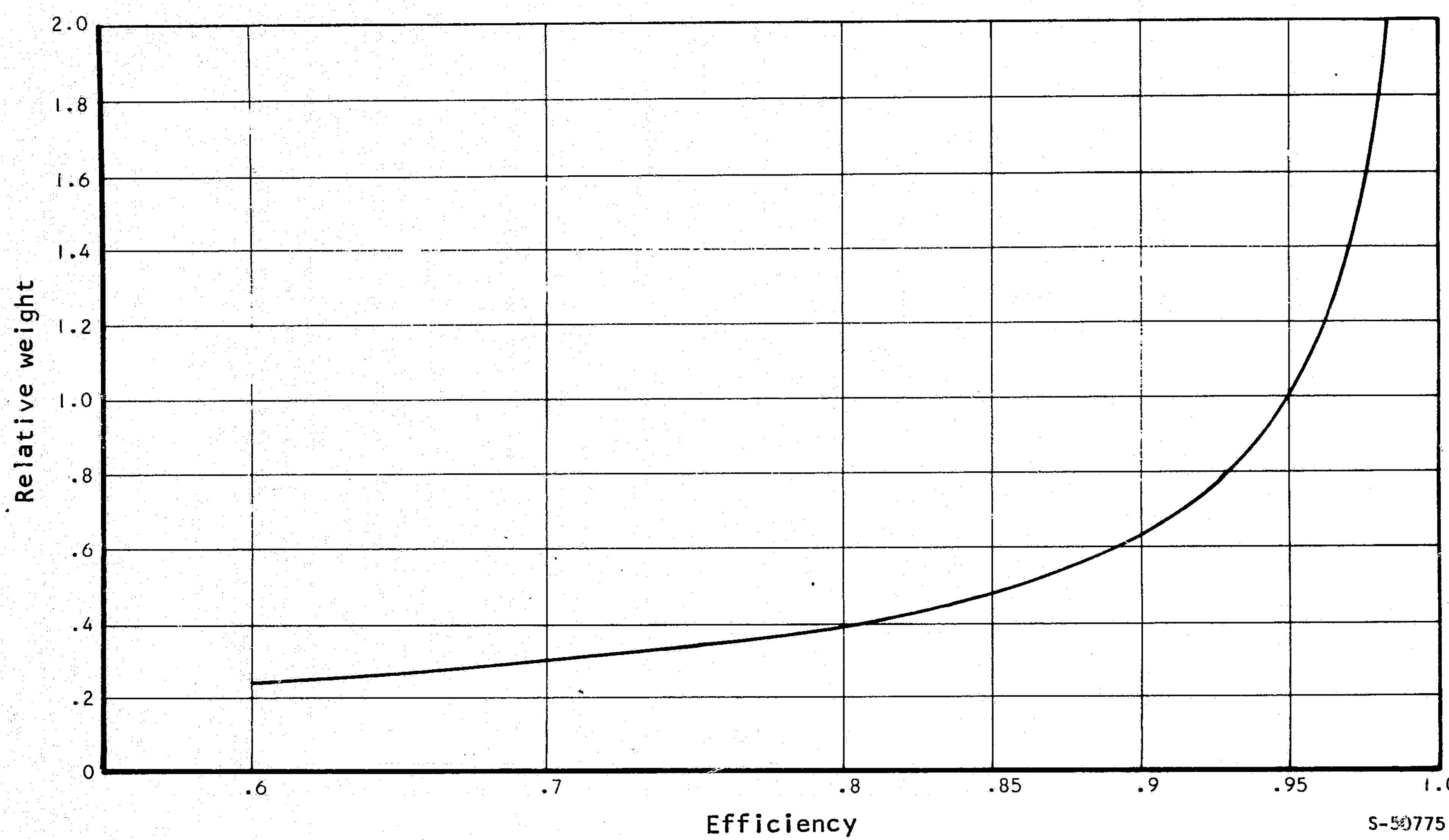
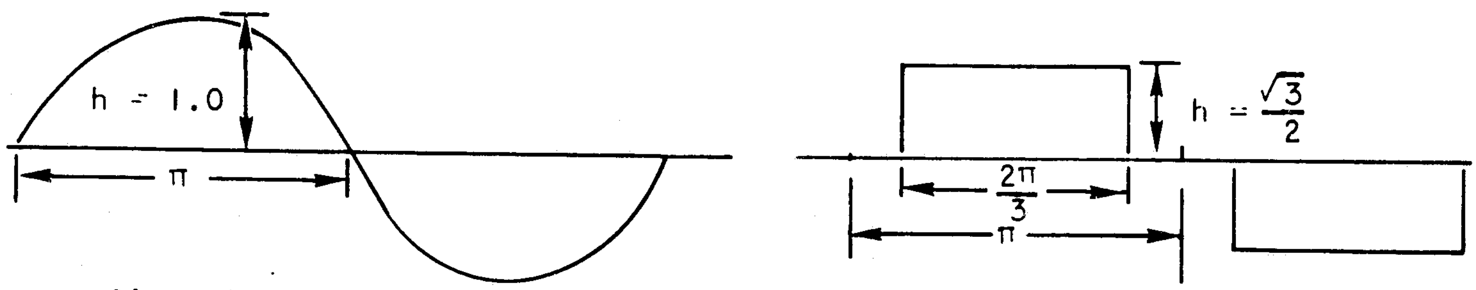


Figure 23. Transformer Weight/Efficiency Relation



a. Sinusoidal voltage wave

b. Stepped voltage wave

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Figure 24. Waveforms of Voltage Applied to a Transformer

Avionics loads.--A list of the avionics equipment in a large subsonic airplane is given in table 6. The avionics equipment primarily uses dc power. Ac power input is, however, required in much of the equipment since: (1) it enables circuit isolation, (2) its voltage can readily be stepped down to several levels if necessary to any desired level, and (3) it is a better regulated source than dc in an aircraft.

Currently, the components in avionic systems are rated in a large variety of voltage levels. For example, in a particular aircraft, 38 different dc voltage supplies and one ac voltage supply are required for the avionic components as shown in table 7. This points out the need for standardization of avionics voltage ratings.

Lighting load.--Aircraft lighting includes exterior lighting (navigation lights, anticollision lights, landing lights, taxi lights, formation lights, approach lights, wing lights, etc.) and interior lighting (instrument lighting, control panel lighting, warning and indicator lighting, floodlighting, general lighting and cabin lighting). Power supplied for lighting in aircraft is generally a basic 115-V, 400-Hz ac, with transformation to 28 or 5 V for incandescent lighting, or to several hundred volts for some of the fluorescent lighting. As mentioned before, lighting consumes approximately 10 percent of the electrical power in a typical large commercial aircraft.

For general purpose interior lighting, the fluorescent lamp is inherently more advantageous than the incandescent lamp. It is much more resistant to vibrations and has longer life and higher efficiency than the incandescent lamps. Also, the heat produced by the fluorescent lamp is small which is an advantage from the view point of the air conditioning system. Fluorescent lamps, however, are not suitable for small area or spot lighting or for high intensity lighting. The RFI effect generated by the fluorescent lamps is also a consideration.

Incandescent lamps for aircraft use are rated in low voltage to give the lamps a reasonable life with heavier lighting elements. Step down transformers are therefore required for incandescent lamps in the conventional 115/200 Vac system.

TABLE 6

AVIONIC LOADS IN A LARGE SUBSONIC AIRPLANE

Name	Input power	Power, VA	Weight, lb
Automatic direction finder	Ac and dc	47	103
Transponder	Dc	73	56
VOR/LOC/ES	Ac and dc	66	195
Distance measuring equipment	Dc	90	99
VHF communication	Ac and dc	240	121
Radio altimeter	Dc	91	93
Marker beacon	Dc	22	18
Audio and public address system	Dc	282	89
Voice recorder	Ac and dc	41	25
HF communication	Ac and dc	1580	319
Selective calling system	Ac and dc	57	11
Air data computer	Ac and dc	130	58
Intercom	Dc	18	63
Weather radar system	Ac and dc	580	166
Navigation radar	Ac and dc	310	255
Misc. communication equipment	Ac and dc		90
Racks, supports and cooling			900

Experimental study by Cooke Engineering, Chicago, (ref. 8) that to reduce the electric field spectrum radiated by fluorescent lamps, the operating frequency should be between 400 and 1000 Hz. Also all sensitive electric cables should be run at right angles to the fluorescent lamp layout. A square waveform of input voltage will improve the efficiency of fluorescent lamps appreciably above that with sinusoidal waveform.

A capacitor ballast weighs less and has less loss than an inductor ballast. High frequency fluorescent lights (400-Hz and above) can use capacitor ballasts. At low frequencies, the capacitor is not usable due to the distortion of the lamp current.

Fluorescent lights can also be operated on dc. In that case, resistance type ballasts are used to control the current. The direction of the current flow through the lamps should be periodically reversed to prevent the reduction

TABLE 7

AVIONIC EQUIPMENT VOLTAGE LEVELS
IN A LARGE TRANSPORT

Regulated	
Voltage	Power
V	W
-2.5	2.5
3.6	1
5	57
6	33
-6	1
-8	7
8.2	1
9	24
10	11
12	10
-12	7
15	17
16	6
18	35
20	67
-20	1
21	148
23	30
24	15
30	6
-30	3
32	3
-40	1
50	1
-50	1
-82	2

Regulated	
Voltage	Power
-90	1
150	1
200	3
6.3 Vac	16
	512

Nonregulated	
Voltage	Power
V	VA
5	10
6	2
12	1
23	2
26	145
27.5	285
35	1470
115	147
230	15
250	10
350	10
460	32
2000	1122
	3251

of light output at the positive end caused by the gradual drift of the mercury to the negative end. The useful lamp life on dc burning is reduced to approximately 80 percent of that on ac burning.

Incandescent lamps will operate on constant voltage supply with any waveform. Fig. 25, taken from ref. 9, shows the variation of watts, lumens and life of incandescent lamps with variation in operating voltage.

Heating loads.--Some aircraft use electrical heating for anti-icing and de-icing. In that case, anti-icing is the most significant heating load in the aircraft. Anti-icing is to maintain the surfaces at a temperature sufficient to prevent the formation of ice. In the de-icing process, ice is allowed to build up to a safe thickness. By applying heat cycles, the ice layer breaks away from the surface and is carried away by air stream. In large modern aircraft, anti-icing is usually performed by hot bleed air from the main engines. Electrical anti-icing is limited to small areas because of the enormous power it demands. Also, aircraft wings are flexible structures, the electrical resistance strips for anti-icing may break when the wings are under motion. On the other hand, bleed air heating requires ducting which would increase the weight of the wing structure. Fuel-air burner is also used for anti-icing purposes.

Windshield de-icing and de-misting are generally done electrically. The power required for de-icing ranges from 500 to 800 W/sq ft (5400 to 8660 W/m^2) while that for demisting ranges from 150 to 300 W/sq ft (1620 to 3240 W/m^2). The windshield area is usually divided into a number of sectors to obtain even heating over the entire area. Ac is usually used for heating because it does not affect the magnetic compass installed in the windscreen area. If dc is used, the wiring arrangement must be capable of avoiding its magnetic interference.

Some control surface actuators are equipped with electric heaters. Operation experience revealed that some actuators failed to operate or slow in action under very low environmental temperature, causing gelling of the lubricants and reduction of bearing clearances on differential contraction. Pitot tube for measuring airspeed also needs electric heater to prevent ice formation at the head.

Air conditioning system also needs electric heaters. Because of the difficulties of air distribution, it is sometimes necessary to augment the aircraft's heating system by introducing elements into air ducts to provide boost heat to raise the temperature level at a particular position in the aircraft.

The electric power demand for galley and passenger comfort systems has increased steadily in commercial aircrafts. Heating loads in galley include ovens, hot cupboards, water heaters, boiling plates, and grill boilers.

Heating loads, in general, do not require closely regulated power. In the Canadair CL-44 airplane, the heating loads are supplied by wild frequency ac with input frequency proportional to the speed of the main engine.

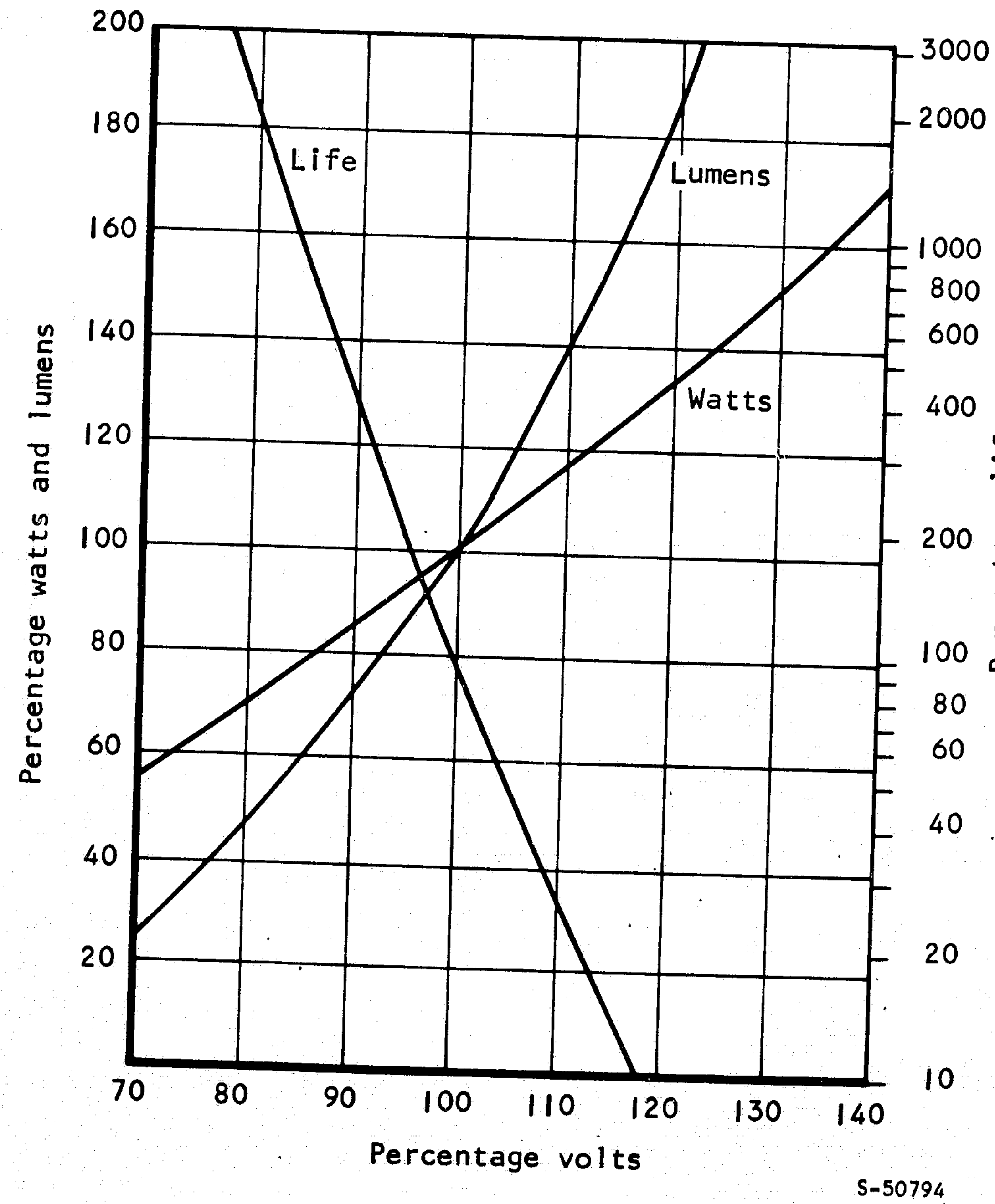


Figure 25. Variation in Watts, Lumens and Life of Lamps with Variation in Voltage (ref. 9)

DISTRIBUTION SUBSYSTEM

Introduction

Many electrical power systems are being used on aircraft today. Among these are (1) 28-Vdc system, (2) 112-Vdc system (four 28-V battery units in series), (3) hybrid system with 28-Vdc and 115/200-V wild frequency 3-phase ac, (4) 115/200-V, 3-phase, 400-Hz constant frequency ac system, and (5) 230/400-V, 400-Hz constant frequency system (in XB-70). The discussion of a distribution system in this report is limited to the constant frequency ac system widely used today on large commercial aircraft.

The constant frequency distribution system on aircraft is in many ways similar to utility power distribution in industry. The vast experience in power distribution gained there may be of value to the aircraft electrical engineer. However, the difference in environmental conditions and power source characteristics requires great care be exercised in applying the practical aspects of power distribution techniques to aircraft conditions.

The function of the aircraft electrical power distribution system is to carry electrical energy from its point of origin to various loads such as fuel booster pumps, avionics, flight control systems, heating equipment, etc. Typically, numerous wire branches spread from the distribution system to these loads; the distribution system starts from generator load buses and also disperses from there. The link from the generator load bus to the generator could properly be called the transmission subsystem or the generator feed. Since much similarity exists in transmission and distribution system design, the entire branch from the generator terminal to the load terminals will be discussed here under the distribution subsystem. A discussion on components such as wires, contactors, relays, switches, circuit breakers, fuses, connectors, terminal strips, transformers, power conversion equipment, control and protection devices, and indicators, will also be included.

The distribution system comprises approximately 25 to 30 percent of the total electrical system weight.

System Configurations

The choice of the system type will generally depend on the following factors:

(1) Number of source buses

(2) Reliability of source buses

(3) Duplication of emergency load functions

(4) Weight, space, and complexity penalties involved in making more than one source bus available for the essential load buses

(5) Degree and type of selectivity available in the generator and bus protection system

For the present discussion, a four-channel generating configuration has been chosen.

Typical distribution system in existing aircraft.-- For a typical four-engine CSCF commercial aircraft, four generators of the same capacity are usually installed, one for each engine. The generators are driven from the main engines via constant speed drives. The capacity of the generator selected is such that the total load demand of the aircraft can be met with only two or three generators operating in parallel although slight load monitoring is usually required.

One or more auxiliary power unit (APU) generators may be installed to supply the ground electrical loads when the main engines are down and to supply bleed air for air conditioning while the engines are idling during taxiing and waiting for takeoff. The capacity of the APU generators is about the same as that of the main generators; however, they may be rated at a higher output because of the extra cooling provided.

The primary electrical power is 3-phase, 400-Hz ac power regulated at 115/200 V. During the flight, this power is supplied by the main engine driven generators; on the ground, it is supplied by either a ground power supply or the aircraft APU (if equipped).

In addition to the primary ac power, the aircraft is also provided with 28-Vac and 28-Vdc power to supply various lights, instruments, and other loads. The single-phase, 28-Vac power is derived from single-phase autotransformers energized by the main ac system through various feeds. The 28-Vdc power is supplied by transformer-rectifier (TR) units which receive input power from the main ac system. To ensure a reliable dc source, more than one TR unit is used. Normally, the TR units are operated in parallel.

In addition to the TR units, a battery is provided as a standby source. In some cases, this source permits starting the electrical system without an external source, supplies backup electrical system control power, and can supply power to minimum navigation, instrument, and communication systems and other 115-Vac equipment needed for safe flight and landing. Where the battery power is the only electrical source, the 115-Vac power is supplied by the battery powered inverter.

The typical distribution system for the four-channel electrical power system is either an all-parallel or a split-parallel system in which each generator is connected to its own three-phase load busbar by a generator circuit breaker. All four generators may be operated in parallel or split into two parallel subsystems, one on each side of the aircraft. This can be accomplished by means of bus tie breakers and split system breakers.

The split-parallel system has the advantage of being completely independent of the two subsystems so that any fault or mismanagement in one subsystem cannot affect the other. This ensures a high reliability for the total generating system. The all-parallel system, however, when designed to meet the one channel out case, gives a better utilization factor. A typical system diagram is shown in fig. 26.

Busbar arrangement.-- A three-phase main busbar is connected to each generator through a generator circuit breaker. The busbars are interconnected by bus tie breakers and split system breakers for parallel operation. Ground power, which can be derived from either the auxiliary power unit or an external supply can be fed to a main ac busbar via a ground power breaker or to the synchronizing bus.

One or more three-phase essential busbars are utilized, normally supplied from the main ac busbars. When the power cannot be obtained from the main bus due to a fault at the main bus, the essential busbars could be supplied from an emergency ac generator driven by the aircraft hydraulic system. The changeover is usually automatic, but it may also be initiated manually, if required.

The essential ac loads supplied by the emergency generator may be divided into two categories:

(1) Aircraft Loads--These loads are essential for handling the aircraft. This power supply will be maintained under all flight conditions until touchdown. When all ac power supplies are out, these loads are supplied by the battery-powered inverter.

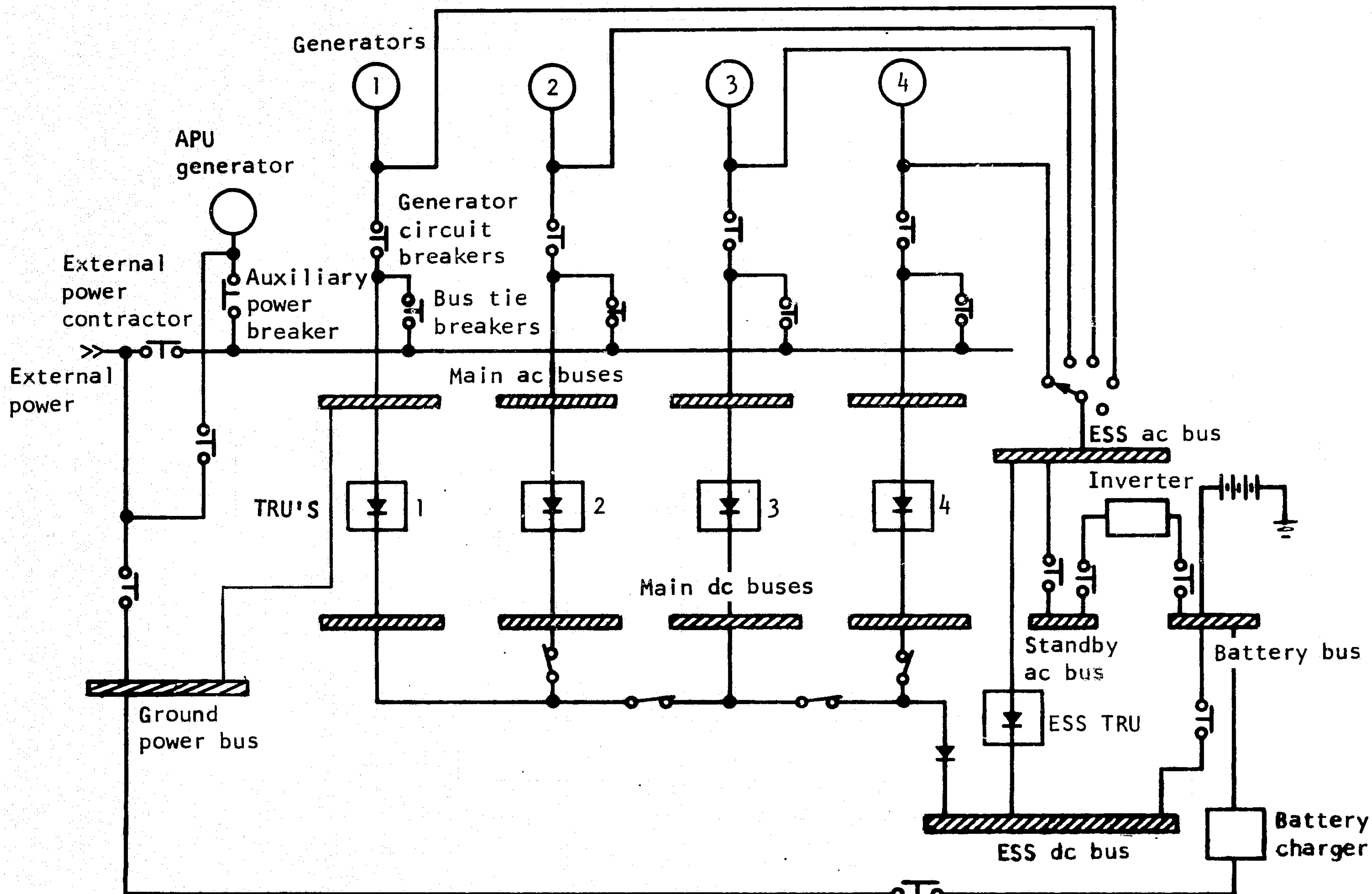
(2) Engine Loads--These are the loads required to maintain the engines. This supply will be discontinued in the event of all engine failure.

One or more dc aircraft loads are supplied from the main dc busbar, depending on the design requirements. During normal operation, this dc power is obtained from the TR units that are energized from the main buses. Normally, the TR units are operated in parallel.

In addition to the main dc busbar, the dc essential busbar is also used to supply the essential dc loads. This busbar can receive power from either the main dc busbar, the ac essential busbar through an essential TR unit, or from the standby battery.

Various lights and instruments require single-phase, 400-Hz, 28 Vac power. A 28-Vac busbar is supplied through an autotransformer from either the main ac busbar or the ac essential busbar.

Depending upon the system configuration, other busbars may be used in the distribution system.


Generators
1
2
3
4
APU generator
Generator circuit breakers
External power contractor
Auxiliary power breaker
Bus tie breakers
External power
Main ac buses
ESS ac bus
Inverter
TRU'S
1
2
3
4
Main dc buses
Standby ac bus
Battery bus
Ground power bus
ESS TRU
Battery charger
ESS dc bus

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(1) Standby 115-Vac Bus--This bus supplies 115-Vac to essential equipment needed for safe flight and landing. During normal operation, it is supplied by the essential ac bus. In the event of loss of essential ac power, the standby 115-Vac bus is supplied by the standby battery through a dc-to-ac inverter.

(2) Ground Power Bus--The 115-Vac and 28-Vdc (not shown in fig. 26) ground service buses provide power to the loads needed during ground operation. The power of the ac ground service bus may be obtained from the external power receptacle or from the APU generator through transfer relays.

(3) Flight Instrument and Radio Buses--These buses provide power to flight instrument and radio equipment are supplied from the primary ac busbars (main ac busbar and essential ac busbar). (This is not shown in fig. (26).

(4) Standby Dc Bus and Battery Bus--These buses provide the standby dc power. They are energized by the essential dc bus during normal operation, but can be transferred to the battery source automatically in the event of loss of essential dc power.

Load transfer.-- Two typical bus arrangements are shown in fig. 27. During normal operation, Scheme A would operate with all the circuit breakers closed. When one generating channel fails, that channel is disconnected from the system and the rest continue to supply the required power. However, Scheme B, a split-parallel system, would operate with the tie breaker open under normal conditions. Thus, the two subsystems are independent. The tie breaker would be closed only during ground operation or a complete failure in one subsystem.

Scheme B has the advantage over Scheme A in that any fault or mismanagement in one subsystem cannot affect the other, thus reducing the possibility of losing the whole generating power due to a single fault. When one generating channel fails in Scheme B, the load is transferred to the healthy channel. When a whole subsystem fails, the tie breaker is closed, and the load in the failed subsystem is supplied by the operating subsystem.

When designed to meet the one-channel out case, however, the advantage which Scheme A has over Scheme B is that the utilization factor is higher. In this four-channel system, Scheme A gives a generator utilization factor of .75, whereas Scheme B gives only .5.

There are alternatives for both Schemes A and B. In Scheme A, two paralleling circuit breakers can be left open under normal operating conditions; thus, there are three independent subsystems.

Scheme B, on the other hand, can operate in such a way that when channel 1 fails, the generator circuit breaker of channel 1 and the paralleling circuit breaker of channel 2 open and the tie breaker closes. Thus, generator

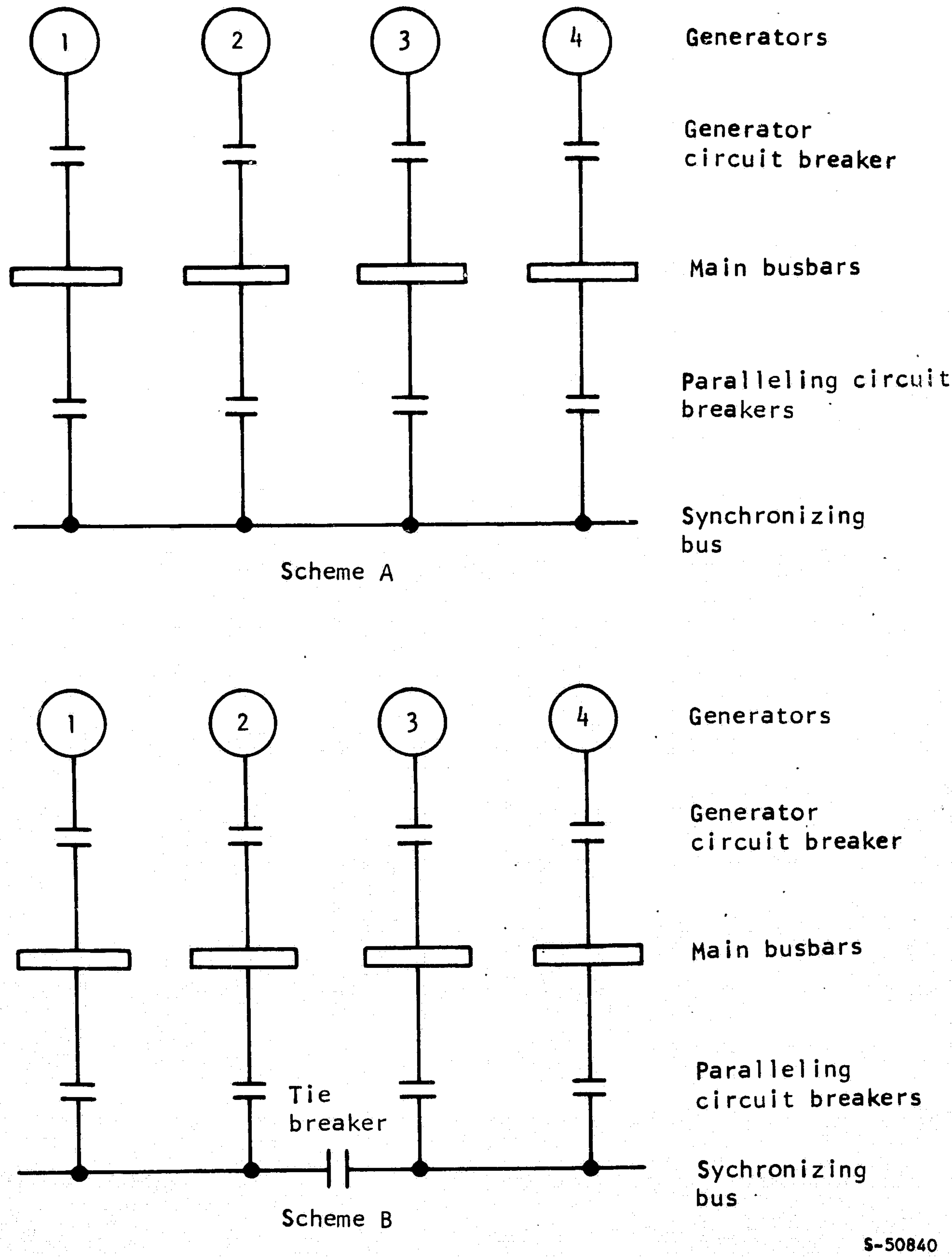

Figure 27. Typical Bus Arrangements

2 will supply its own load, and the load on channel I will be supplied by generators 3 and 4.

Surface deicing in the aircraft requires a great amount of power. Since this power is needed only occasionally, the load is usually split into two parts and supplied by the two subsystems, respectively, in a split-parallel configuration. This limits the amount of power to be transferred, and uses the installed generating capacity more efficiently.

When one generating channel fails, the load transfer can be accomplished by an arrangement shown in fig. 28. Asuming generator 4 fails, contactor B can be switched from right to left, transferring the deicing load from generator 4 to generator 2. The general loads on busbar 4 will be supplied by generator 3.

There are many different types of load transfer arrangements, the choice of which depends on such factors as the reliability requirement, the generator utilization factor, and the characteristics of the loads.

Distribution Components and Parametric Data

Aircraft cables.-- The wiring installed on large transport aircraft makes up for about 70 percent of the electrical system weight (generation and distribution only, not including load equipment). It is the heaviest single item in the entire electrical system and therefore attracts special attention in the continuous effort of weight reduction. In view of this, the importance of selecting the optimum wires (type and gage) for the various application is evident. Aside from the basic requirement of carrying current without excessive voltage drop and heating, number of other factors must be considered. Starting once more with weight, these factors are discussed in the following paragraphs.

Weight: A considerable portion (40 to 70 percent) of the total installed wire weight is contributed by cables of size 20 or smaller. The ratio of the insulation weight to the conductor weight increases as the cable size decreases. The weight of copper is about equal to the weight of insulation for size 18 wire. For wires smaller than size 18 the weight reduction in insulation is, therefore, more meaningful than the weight reduction in the conductor.

Flexibility: It is desirable that the cables remain flexible over the full range (-50° to $+100^{\circ}$C) of ambient temperature. In areas adjacent to the engine, the maximum temperature may reach 300°C. The cable insulation should not become too soft or too brittle within the temperature range to which it is exposed.

In present aircraft there are usually many wires in a harness, and many wires are required to terminate at miniature or subminiature high-density terminal blocks. These wires plus the limited access to installation areas, tend to require more flexible cable.

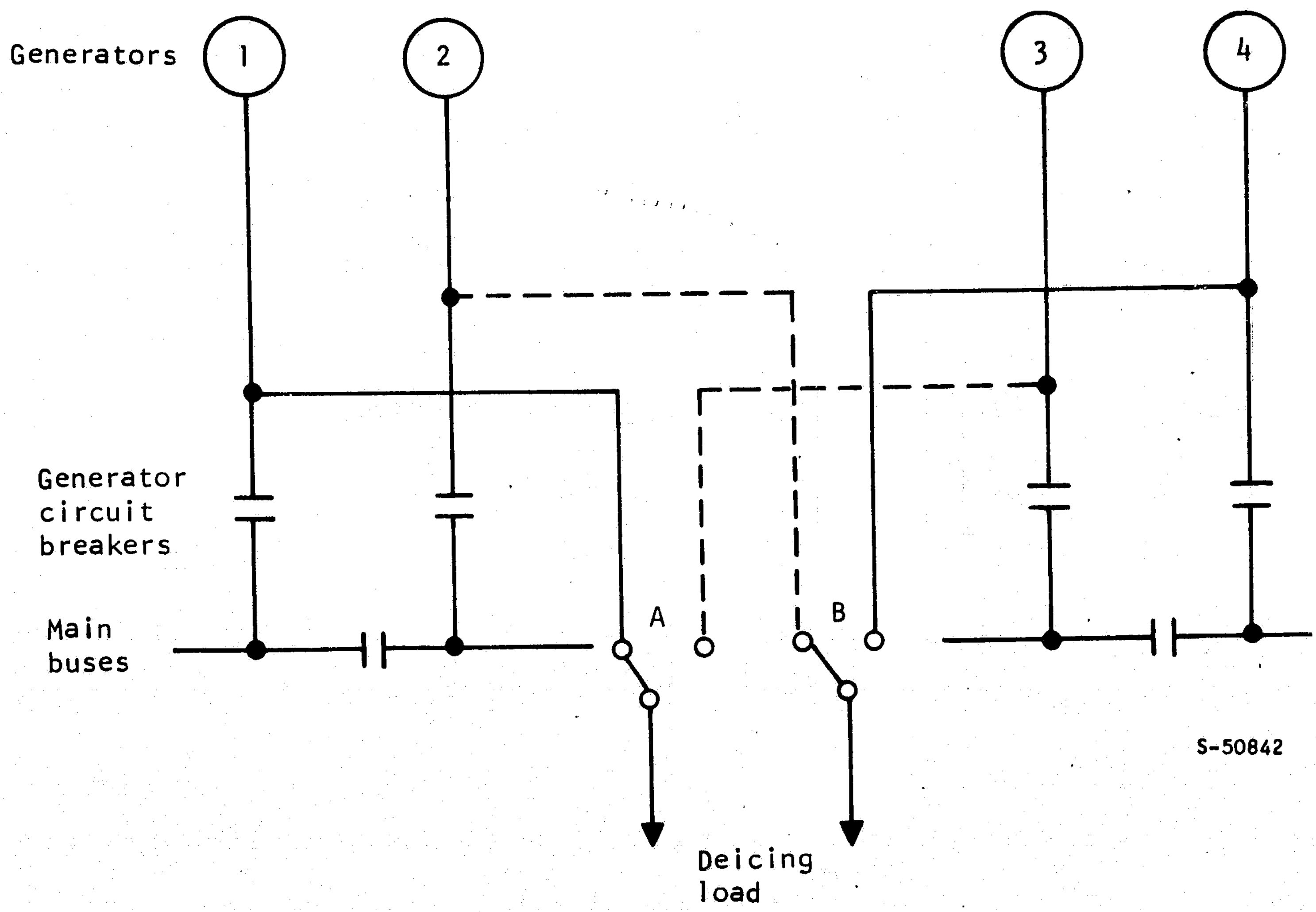

Figure 28. Load Transfer Arrangement

Abrasion: During installation and maintenance, abrasion is almost unavoidable due to either installation design weaknesses or insufficient care by maintenance personnel. In flight, abrasion may occur because of vibration.

To minimize this abrasion hazard, every conceivable precaution should be considered. The cable should be sufficiently resistant to abrasion in the first place to prevent the wear and tear from causing a short circuit to ground. For comparing cables of different construction, an accurate method of determining the abrasion resistance of the cable is highly desirable. Unfortunately, this has been difficult as the repeatability of the standard abrasion test is very poor, unless the wide tolerances inherent in the test procedure are acceptable.

Aircraft and cable manufacturers have been seeking various forms of abrasion tests for evaluation purposes, including:

(1) Scrape--Simulates the action of a metal edge moving relative to the cable insulation

(2) Cut through--Simulates the action of pressure from a metal edge on the cable without relative motion

(3) Notch--Determines the tear strength of the cable and provides a measure of toughness

Successful means of determining cable abrasion resistance will be very valuable in improving cable characteristics.

Contamination: In many locations within the aircraft, wiring may come in contact with and be contaminated by various fluids such as fuels, lubrication oils, coolants, anti-icing sprays, and hydraulic fluids. In some cases, the contamination may be accompanied by elevated temperatures, enhancing the destructive effect. To avoid this, it is desirable that all general purpose aircraft cables should have appropriate contamination resistance against all those fluids known to be used in aircraft.

Since synthetic lubricants and kerosene fuels are widely used in contemporary aircraft, insulating materials such as nylon that cannot be contaminated by these fluids should be used in outer sheaths.

Fire resistance: Deficiency in aircraft wiring is known to be one of the major sources of fire hazards. Various causes which would induce fire in wiring are:

(1) Circuit overload due to faulty electrical equipment

(2) Inadvertent shorted wires

(3) Wet wire fire

A number of fire incidents on aircraft have been identified as "wet wire fires." Peculiar of those incidents was the inability to attribute the cause of wire fires to particular circumstances such as circuit overloads due to faulty electrical equipment or inadvertent shorted wires. In some cases, circuit protection devices proved ineffective or did not respond until after smoke or fire had started.

This initiated closer investigation. Major airframe manufacturers conducted laboratory tests, simulating conditions which were believed to occur in the aircraft. Two conditions found to cause the hazard were damaged insulation and the presence of moisture. A third condition was the necessity for energizing the wire bundle. Since moisture is essential to the process, this hazard was identified as "wet wire fire."

So far, wet wire fires have occurred on general purpose aircraft wiring only. Since about 80 percent of all wiring in present aircraft is made up of such wire, an improved general purpose wire is needed to eliminate or minimize the potential danger of wet wire fires.

In the engine areas, some circuits are required to operate during and after a fire. Therefore, fire resistant cables must be used that can operate satisfactorily after being subjected to a temperature of 1100°C for 5 min without creating any additional fire hazard. In other areas, cables must at least be fire resistant and not support combustion after the source of ignition has been removed.

Mechanical strength: All aircraft wirings are subject to being pulled through ducts, conduits, on terminal lugs or by accident during installation and maintenance. They are also subject to forces due to vibration. Without sacrificing other cable characteristics, the tensile strength of the cable must be as high as practical, particularly for the smaller cables. Although copper is a good conductor, it has poor tensile strength and the ordinary type of insulation offers little assistance. For this reason a 22-gage copper cable was the smallest size used in aircraft. The new cadmium-chrome-copper alloy, however, has increased the tensile strength so much that wires as small as 26 gage are presently installed in aircraft.

Additional mechanical strengths such as resistances to deformation and bending are also desirable. These resistances are primarily a function of the type of insulating material. For example, the cushioning effect of glass braid insulation can enhance deformation resistance of cables.

Current overload performance: All cables in aircraft require protection against overload. The protective device used, whether fuse or circuit breaker, must be compatible with the thermal characteristic of the cable to prevent serious cable degradation which would reduce its life below that of the aircraft. Some equipment have very high inrush-current characteristics. To prevent these transient currents from tripping the protective device, a larger cable must be used with a current carrying capacity exceeding the continuous load current.

Because the protection device has a manufacturing tolerance on its trip characteristic (for a typical circuit breaker the minimum and maximum trip values are 110 and 138 percent of the nominal value respectively) and is available only in a certain range, it is very difficult in practice to use the cable at its stated rating. If, however, the cable were allowed to carry a current in excess of the specification rating for a limited time, the restriction on its rating could be largely overcome.

Publications of complete short-term ratings for standard aircraft cables cannot be found. Test procedures are being evaluated, however, to provide the required information for matching circuit breaker characteristics more accurately, to the cable, and accomplish maximum savings in cable size and weight.

Conductor materials: Copper and aluminum have both been used as conductive materials in aircraft cables. The high conductivity of copper together with its ability to be soldered or crimped easily has made copper the preferred conductor in the past despite its disadvantage of high density. Aluminum has a volumetric resistivity of 1.6 times that of copper, however, the ratio of densities of aluminum to copper is only 0.3. Therefore, the weight of an aluminum wire is one half that of a copper wire of the same resistance. Because of its low mechanical strength, however, its use is limited to wires of size 8 or larger. The aluminum cable also has a termination problem. Better termination techniques must be developed before extensive application of aluminum wires on aircraft can be realized.

Since many circuits in aircraft carry only a small amount of current, conductivity is not a limiting factor. A suitable conductor should have, however, an acceptable combination of both electrical and mechanical properties. Copper-dominated conductors which have been used for aircraft cables for many years include tinned copper, nickel-clad copper, and silvered copper. Noncopper conductors which have been used include aluminum conductors of larger sizes, stainless steel conductors, and conductors made of other alloys. Recent trends indicate that copper-base and aluminum-base alloys that give a satisfactory balance of mechanical and electrical properties include chromium-copper and the ternary alloy cadmium-chromium-copper. The chromium bearing alloys have very attractive mechanical properties.

The feasibility of using aluminum alloys such as aluminum-magnesium-silicon as conductor appears attractive and is being investigated. As with the aluminum conductor, aluminum alloys may not be satisfactory for use in small wire sizes and their applicability may depend largely on the development of successful termination techniques.

Cable insulation: Most insulating materials used on aircraft cables are synthesized from basic materials by a chemical process involving polymerization. This process is usually accomplished by applying heat, pressure, or a certain catalyst to the chemical reaction under strictly controlled conditions. Synthetic cable insulations used include polyvinyl chloride (PVC), polytetrafluoroethylene (PTFE), teflon, nylon, kynar, and cross-linked polyalkylene.

The only nonsynthetic insulating materials used are glass, asbestos, and rubber. Before they can be used as insulation, however, some modification process is normally required to change their natural state.

Cable insulation usually consists of several layers. The first layer, from the conductor outward, may be a conductor wrapping made of inorganic material. The next layer is the main insulation, and is usually made of PVC or teflon. Then glass braid may be used for the next layer, if desired, and an outermost covering is made of either nylon or some other composite material.

Table 8 shows the properties of the most commonly used insulation materials.

Current carrying capacity: The current carrying capacity of the cable at given ambient conditions is determined by the maximum allowable temperature of the cable, which is in turn, dependent on the type of cable insulation and its heat dissipation capability. For continuous loads, the cable temperature must remain within certain limits to avoid excessive aging, discoloration, and insulation deterioration. Therefore, for each type of cable, a maximum allowable current at a given ambient temperature is specified for each cable size. Table 10 shows the current carrying capacity together with other data for aircraft wires and cables for the present 115/200 V, 400-Hz system in accordance with MIL-W-5086 and MIL-W-5088. It shows two current values for each size of cable, one for single conductors in free air and the other for bundled wires (15 or more wires per bundle).

The current ratings for the bundled case are lower because the heat dissipation capability of the cable is reduced. For smaller bundles, the allowable percentage of total current may be increased as the bundle approaches the single wire condition.

The current ratings shown in table 9 apply to general purpose wires for maximum wire temperature of 105°C with a maximum ambient temperature of 60°C. They also apply to wires that comply with MIL-W-7139, Class 1 with a maximum conductor temperature of 200°C and a maximum ambient of 155°C and to wires that comply with MIL-W-7139, Class 2, with a maximum conductor temperature of 260°C and a maximum ambient temperature of 215°C.

Fig. 29 shows the permissible current density and the weight per kVA per unit length as a function of the conductor cross-sectional area for various size wires. As a conductor increases in size, the ratio of the surface area to the enclosed volume decreases. Since the ability to dissipate heat is a function of the surface area, the permissible current density in the smaller size wire is greater than that in the larger size. Since the weight per kVA per unit length of the conductor is inversely proportional to the permissible current density, it will increase as the cross-sectional area of the conductor increases. In other words, the smaller size wires can transmit more power for a given weight of wire than the larger sizes.

TABLE 8

PROPERTIES OF AIRCRAFT CABLE INSULATIONS

Properties	Polytetrafluoroethylene	Polyethylene	Cellular polyethylene
Specific gravity	2.1 to 2.3	0.92	0.50
Volume resistivity, ohm-cm (50 percent relative humidity and 23°C)	10^{15}	$> 10^{15}$	- -
Dielectric strength, short-time 1/8 in. (3.17 mm) thickness	480 V/mil (18,900 V/mm)	460 V/mil (18,100 V/mm)	220 V/mil (8,650 V/mm)
Dielectric strength step-by-step 1/8 in. thickness (3.17mm)	430 V/mil (16,900 V/mm)	420 V/mil (16,500 V/mm)	- -
Dielectric constant, 10^6 cycles	2.0	2.26	1.5
Dissipation (power) factor, 10^6 cycles	.002 to .005	<.0005	.0004
Effect of strong acids	None	Attacked by oxidizing agents	- -
Effect of strong alkalies	None	Resistant	- -

TABLE 9

PROPERTIES OF AIRCRAFT CABLE (COPPER)
(MIL-W-5086)
(ENGLISH UNIT)

AN size	$20^{\circ}C$ ohms per 1000 ft	Max. Amp. $60^{\circ}C$ ambient		Cir MIL area	Max. dia. cable, in.	Max. dia. conductor, in.	Weight finished cable, lb/1000 ft
		Bundled	Single in free air				
22	15.52	5	--	755	.075 ±.005	.033	4.7
20	9.7	7.5	11	1,200	.085 ±.005	.041	6.8
18	6.08	10	16	1,909	.095 ±.005	.052	9.5
16	4.76	13	22	2,409	.015 ±.005	.061	11.9
14	2.99	17	32	3,830	.125 ±.007	.076	18.3
12	1.88	23	41	6,088	.143 ±.007	.096	26.0
10	1.16	33	55	9,880	.189 ±.007	.128	44.0
8	.70	46	73	16,864	.240 ±.007	.176	70.0
6	.436	60	101	26,813	.293 ±.007	.218	110.0
4	.274	80	135	42,613	.355 ±.010	.272	165.0
2	.179	100	181	66,832	.425 ±.010	.345	250.0
1	.144	125	211	82,108	.470 ±.010	.384	305.0
0	.114	150	245	105,022	.525 ±.015	.432	400.0
00	.090	175	283	133,665	.590 ±.015	.490	500.0
000	.072	200	328	167,332	.650 ±.015	.548	620.0
0000	.057	225	380	211,954	.720 ±.015	.615	785.0

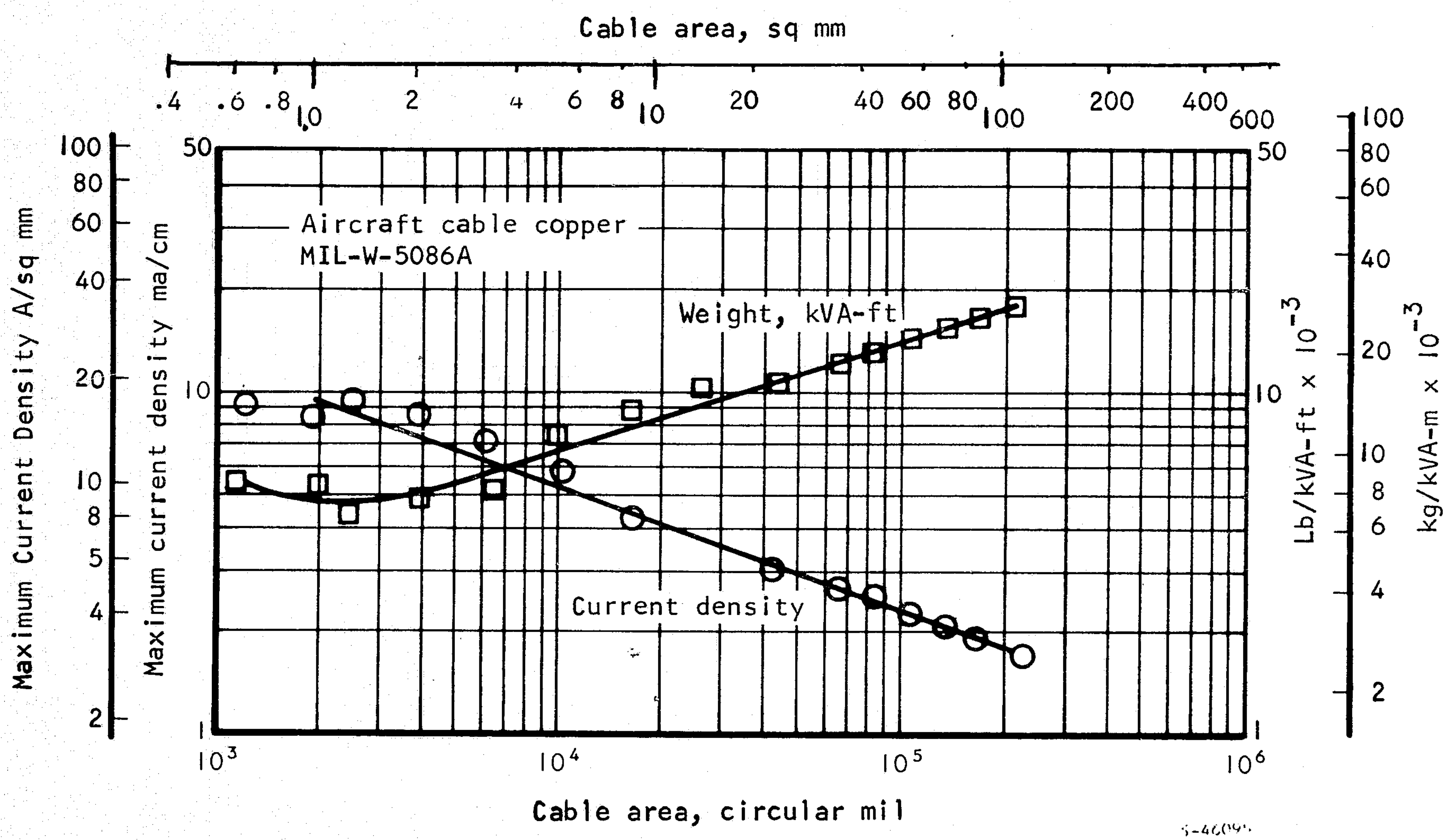

Figure 29. Conductor Maximum Current Density and Weight/kva-ft vs Wire Area

When the ambient temperature is above the maximum specified, the current carrying capacity is reduced. This is because (1) the high ambient reduces the permissible temperature rise and (2) it increases the wire resistance and, therefore, the losses. Altitude also affects the current carrying capacity. For example, the resistance of a copper conductor at 350°C is 2.25 times that at 25°C. Based on the resistance drop, at 350°C, it would take 2.25 times as much copper to carry the same current as at 25°C. At high altitude, the heat dissipation capability is reduced due to the reduced air density; hence, the current carrying capacity is degraded. The Naval Research Laboratory has conducted experiments and has established that for wire sizes 20 through 0; the relative values necessary to produce the same conductor temperature rise are as shown in the table 10.

TABLE 10

ALTITUDE DERATING OF CURRENT-CARRYING CAPACITY

Altitude	Relative current, percent
0	100
20,000 ft (6,100 m)	93 ±2
60,000 ft (18,300m)	82 ±2

Voltage drop: Voltage drop is another factor that governs the selection of the cable size for a given application. According to MIL-W-5088, the continuous value of the maximum allowable voltage drop of the 115-V system is 4V. That means the total impedance of the cable and the ground return in the circuit should not drop the voltage more than 4 V between the point of voltage regulation and the load. This requirement limits the length of any wire size. Fig.30 shows the maximum lengths of cable which can be used for each size for two conditions, the single conductor and the bundled. In present commercial aircraft, wire lengths of over 100 ft (30.5 m) are common and voltage drop considerations are usually the governing factor in selecting a smaller cable.

In a 115/200 V, 400 Hz system, the rated generator terminal voltage is 120 V. The allowable drop from the generator terminal to the point of regulation (main distribution bus) is therefore 5 V. The power transmission limits of aircraft cables in a 115/200 V, 400 Hz system are shown in fig. 31 when a 5 V allowable drop is assumed. The wire configuration is also shown in that figure. There are two curves for each size of cable. One curve, labeled "in air," refers to a wire configuration with sufficient clearance between wires that it can be considered equivalent to a single wire in air. The other curve is for bundled wires. The curved portion of each plot represents the effect of the voltage drop limitation on the power carrying capacity; the straight portion represents the temperature limitation. It can be seen that a larger wire can carry power at its rated capacity a longer distance than a smaller wire.

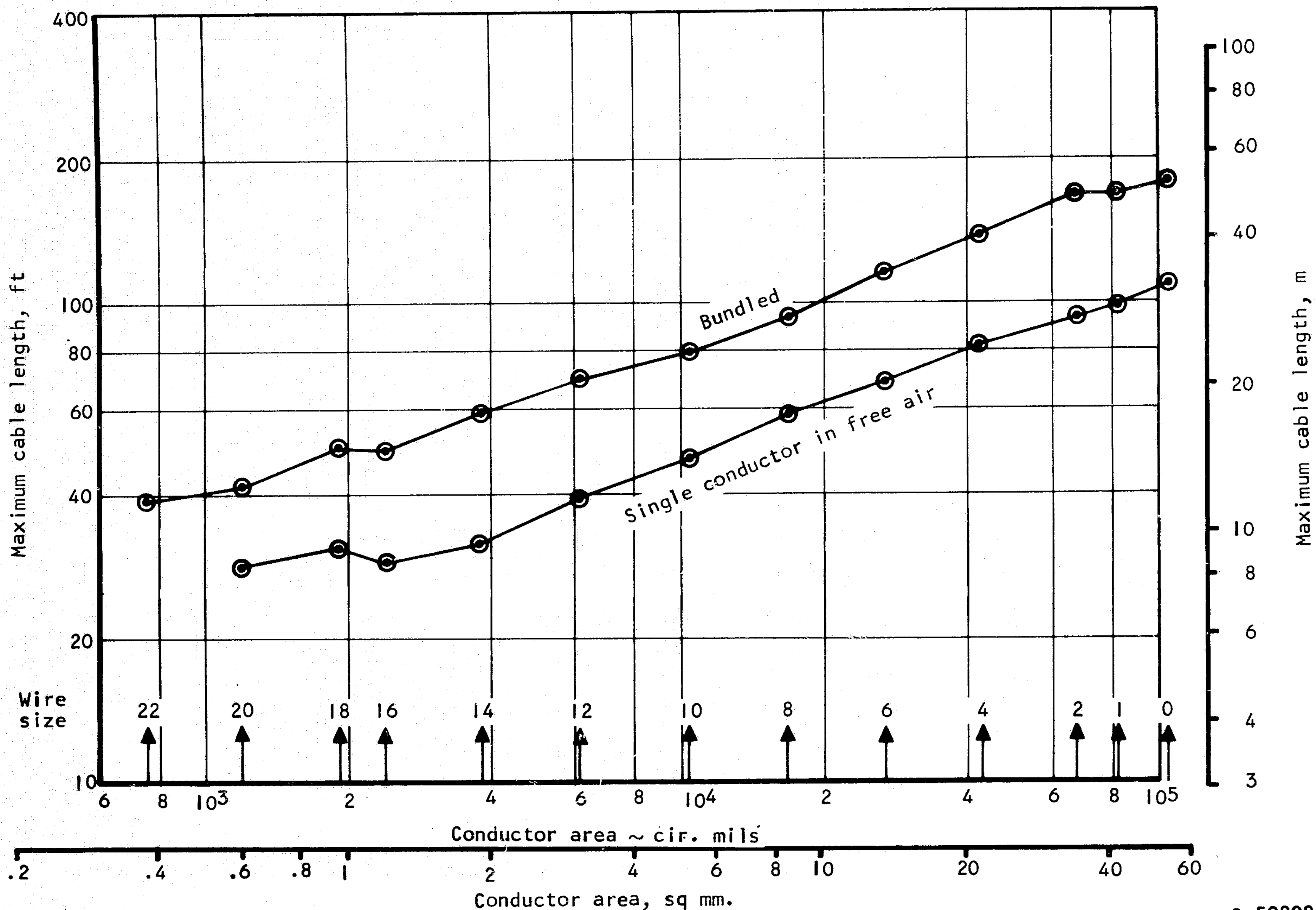

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Figure 30. Conductor Cross Sectional Area vs Maximum Cable Length for an Allowable Voltage Drop of 4 Volts

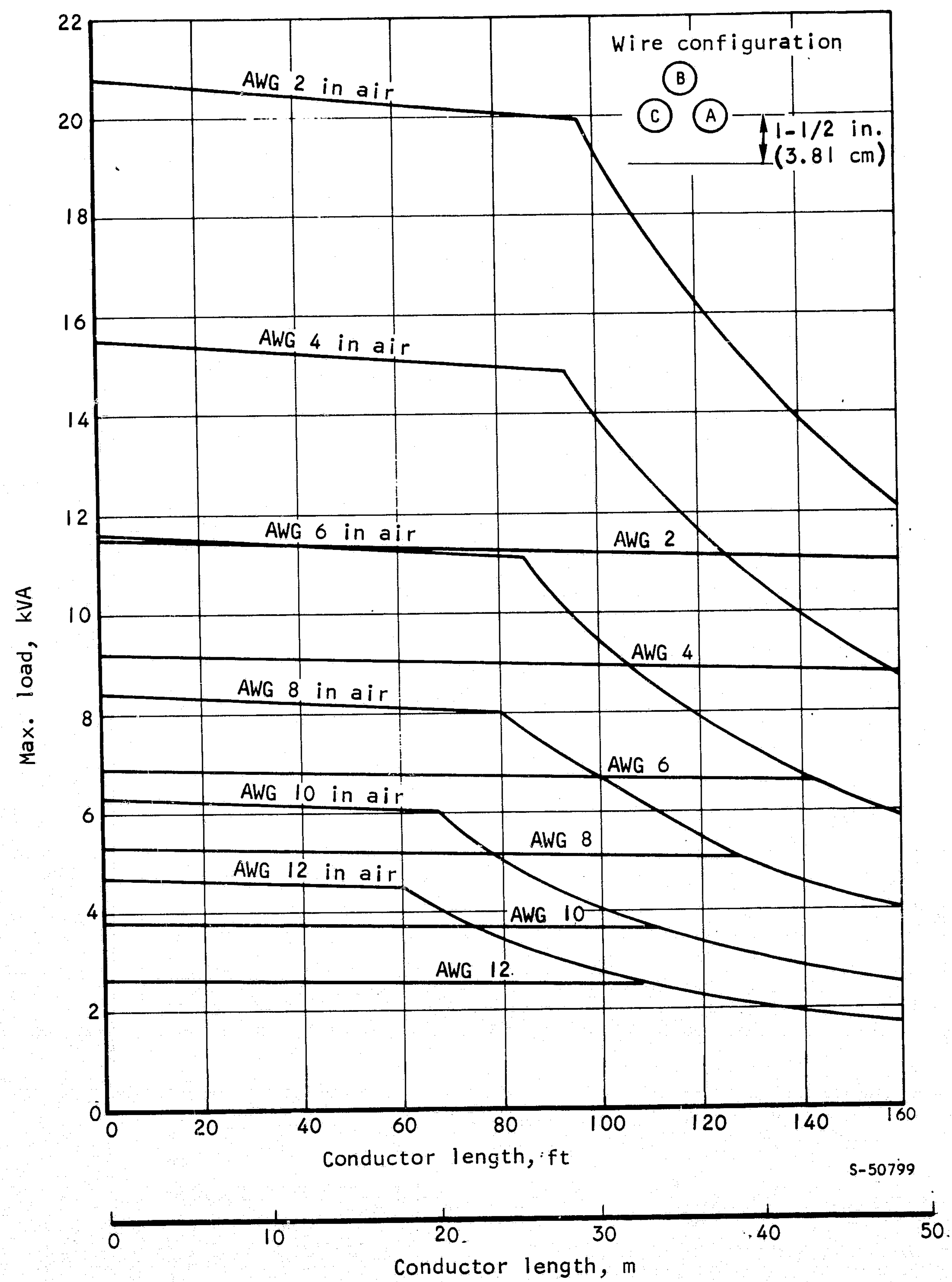


Figure 31. Power Transmission Limits of Aircraft Cables

Comparison of distribution wire weight: To compare the main feeder weight of some typical systems, the weight per 100 ft (or per meter)of feeder cables for 115-V, 400-Hz, 3-phase ac systems of various generating capacities are calculated and plotted in fig. 32. The upper curve is for a 3-wire system while the lower one is for a 6-wire system (two wires for each phase). All conductors are considered in free air, and the current carrying capacity specified in MIL-W-5088C (ASG) is used. A maximum allowable voltage drop of 5 V is selected. It is seen that the 6-wire system results in savings at large kVA ratings since, as mentioned previously, the small wires for a given weight have greater power carrying capability than the larger sizes. This is because with a larger surface-area to cross-section-area ratio, the smaller wire operates at a lower temperature for the same current density. (At low kVA with small wires, voltage drop rather than current carrying capacity, is the limitation.)

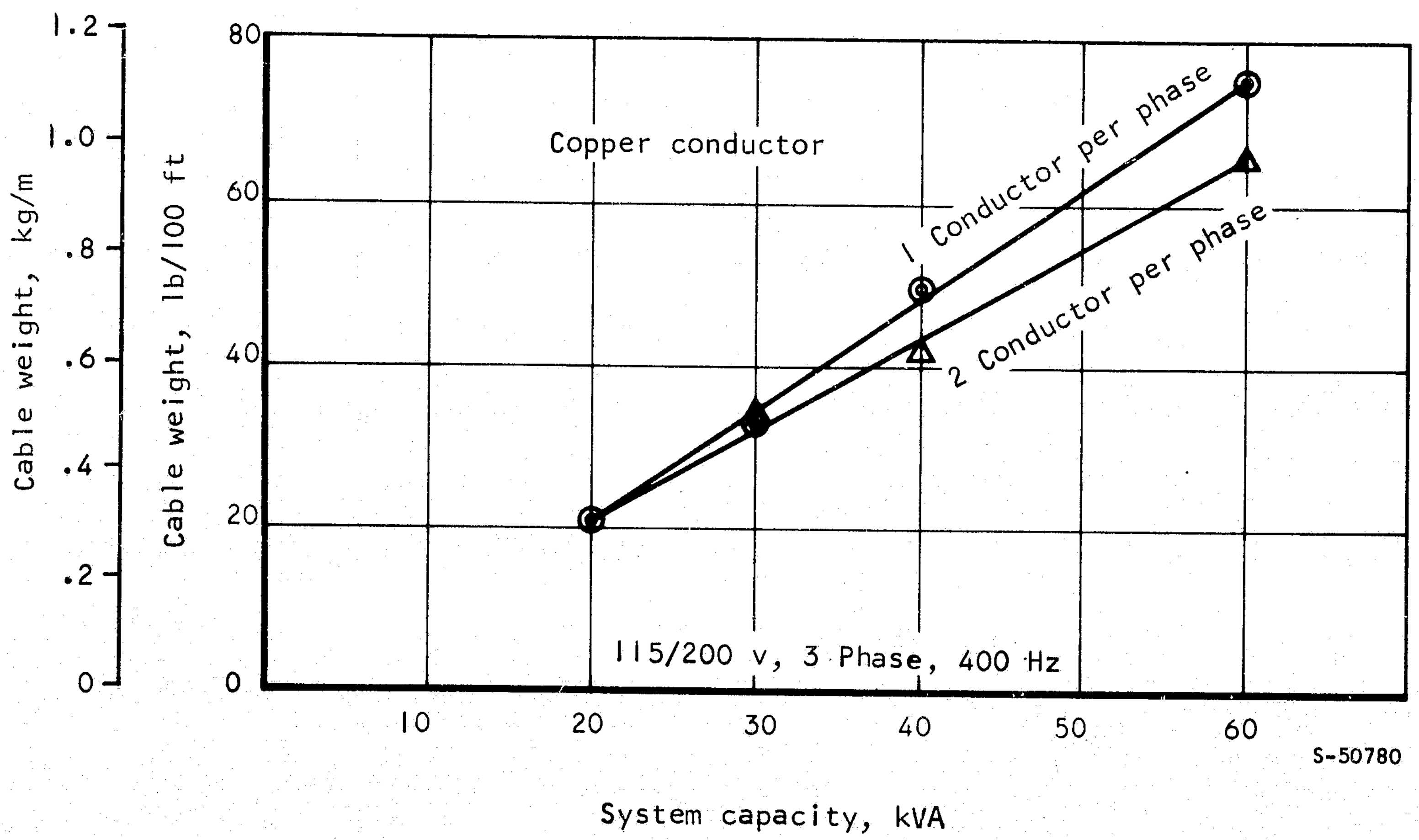


Figure 32. Copper Wire Weight vs System Capacity

Fig. 33 shows cable weight for system using aluminum wires. Although the weight reduction is attractive, factors such as termination and mechanical strength must also be considered.

The dc system may require a return wire, as in the case of supersonic aircraft with titanium structure. With corresponding system voltages ($V_{dc}=2V_{ph,ac}$) more power can be transmitted by a dc system than with a 400-Hz, 3-phase ac system since the ac system has power factor and reactance drop disadvantages.

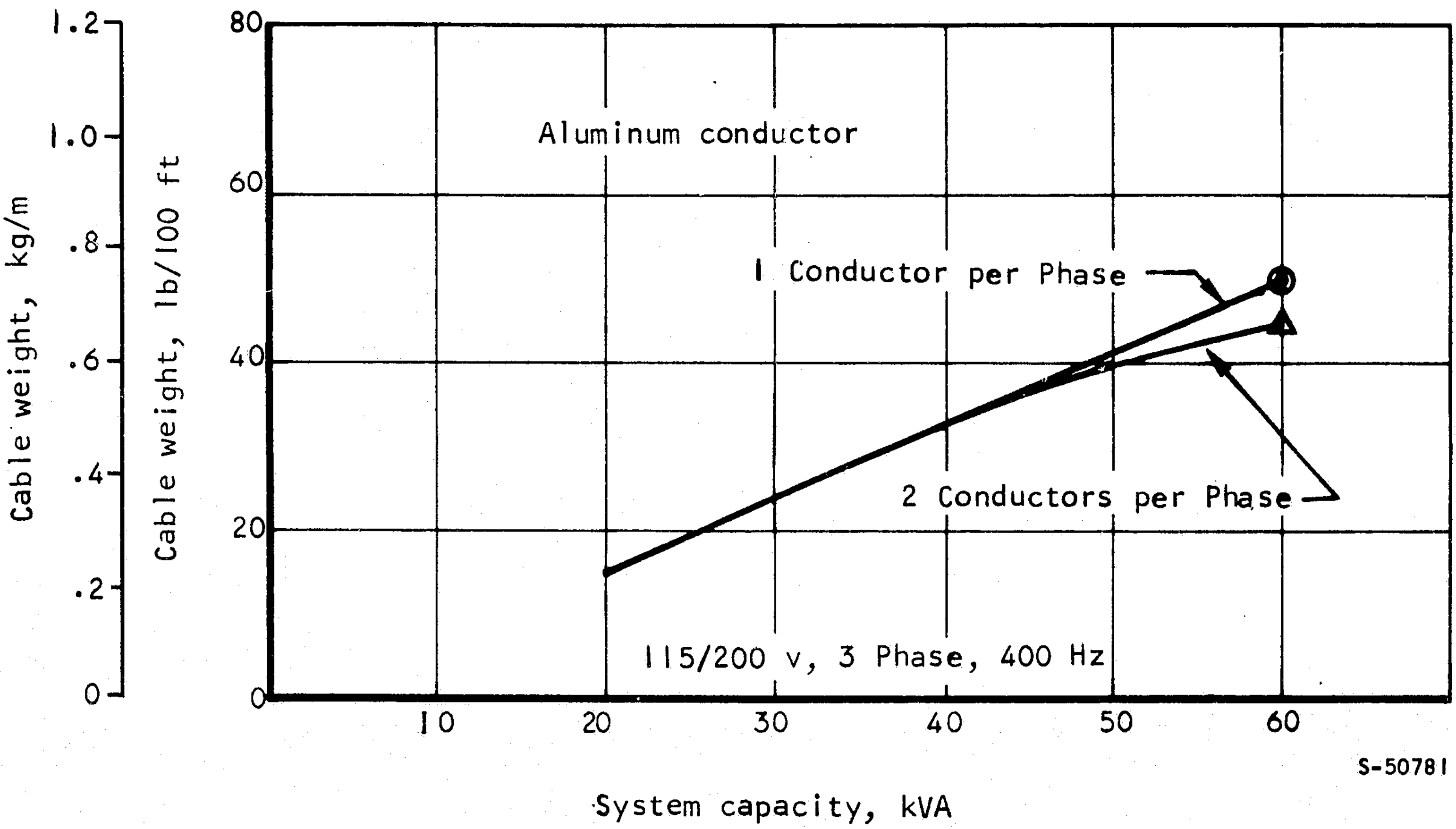


Figure 33. Aluminum Wire Weight vs System Capacity

Fig. 34 indicates that the cable weight of a 112-Vdc system using a return wire is about two times that of the ac system for system capacity in the range of 20 to 60 kVA. If frame is used as return, the cable weight of a dc system may come down to as low as 60 percent that of the ac system. This is because the allowable wire voltage drop in a frame return system is almost double that of a 2-wire system since the voltage drop in the frame is relatively small.

Installation requirements: Installations may vary with the size and the type of the aircraft involved as the amount of space available and the electrical power system configurations are governing factors. Installations may also vary between different manufacturers. Basic installation guidelines are given below.

(1) The risk of mechanical damage to the cable should be reduced to a minimum by proper and careful routing. The cable should be kept away from all moving parts such as control cables, control surfaces, and linkages, etc. Areas which can be easily contaminated should be avoided whenever possible.

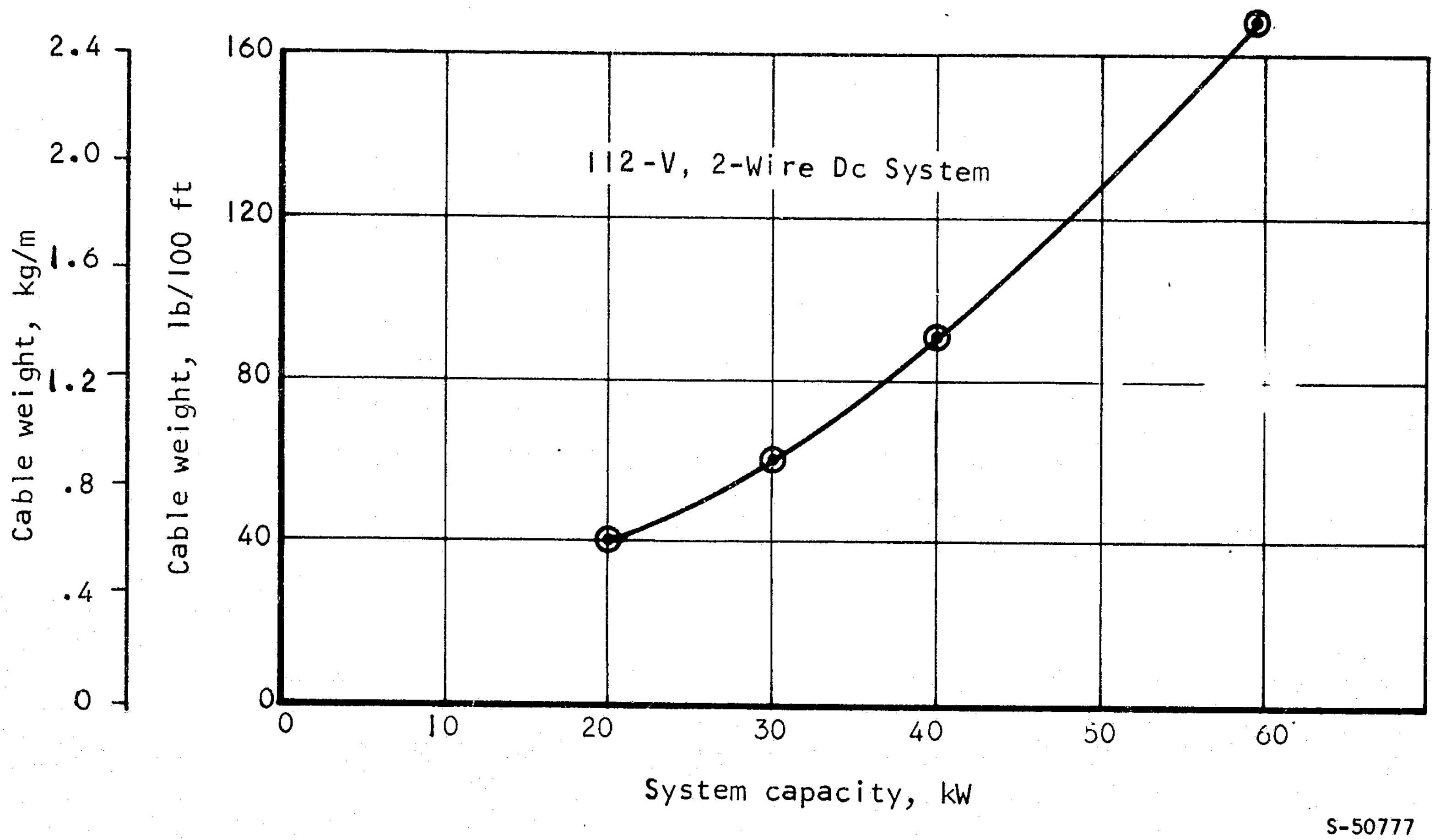


Figure 34. Wire Weight in Dc System vs System Capacity

(2) Cables should be properly supported to prevent chafing and rubbing, beating under vibration, axial movement, mechanical strain, and any possible physical interference with other equipment.

(3) When leaving or entering pressurized areas, cables should be effectively sealed against air leakage. It is preferable not to cut cables when passing them through pressure walls.

(4) Some cables require more mechanical protection than others. This additional protection has been provided by using ducting, conduit, or sleeving; this, however, adds considerably to the weight and cost of electrical installation.

Contactors.-- Contactors are electromechanically or mechanically operated devices for making and breaking circuits. Although contactors are sometimes called circuit breakers, circuit breakers are never called contactors. Circuit breakers differ from contactors in that they interrupt a current not only under normal operating conditions but also under abnormal conditions such as a system fault. A circuit breaker may also have a built-in automatic tripping device which a contactor does not have.

In the typical distribution arrangement, two contactors are used per generating channel: a generator contactor and a bus tie contactor (GCB and BTB are abbreviations for generator circuit breaker and bus tie breaker respectively). If an APU is installed, an external power contactor (EPC) and an auxiliary power unit breaker (APB) are also used, making a total of 10 to 12 identical three-phase contactors in the system.

If installed channel capacity is 30, 40, or 60 kVA, the steady-state current for the contacts will be 83, 111, or 167 A respectively. The contacts are required to break a fault current of 5000 A. There are also several sets of auxiliary contacts, both normally open (N.O.) and normally closed, (N.C.) that are available for circuit interlock functions, current transformer loop sharing, and breaker state indication. For this application, the contactors are of the mechanically or magnetically latched type in which both closing and tripping are effected by momentary current pulses in separate operating coils. A source independent of the main generating system such as the permanent magnet generator supplies the 28-Vdc control power for the coil. Specification MIL-C-8379A requires that a 3-pole, 208 V, 120-A, 400-Hz circuit breaker have operating times no longer than 10 msec for closing and 10 msec for interrupting. The breaker will occupy a volume not more than approximately 4 x 4 x 8 in. and weigh less than 5.5 lb (2.5 kg).

Relays.-- Relays are used for various power and control functions throughout the aircraft. They come in ratings such as those used for dry circuit applications and have power handling capabilities of over 100 A. Relays may be installed anywhere in the aircraft and may therefore be subjected to a wide varying range of environmental conditions. Environmental requirements include resistance to ambient temperature variations, pressure variations, shock, vibration, and corrosion. Typical environmental conditions for present commercial jet aircraft are:

(1) Temperature range from -65^{0} to $125^{0}C$

(2) Pressure down to 1.3 in. Hg (corresponding to 70,000 ft altitude)

(3) Shock pulses of 15 to 50 g acceleration with minimum frequency of 2000 Hz near the engine

(4) 10 g acceleration with frequencies up to 2000 Hz remote from the engines

Relay selection: Aircraft relays should generally provide the following characteristics:

(1) Minimum weight consistent with reliable operation

(2) Sufficient closing and opening forces to ensure proper operation during acceleration or vibration

(3) Satisfactory operation in every respect between 75 and 125 percent of nominal system voltage

(4) No effect upon relay rating with respect to change of position

(5) Nonwelding alloy contacts or hammer action to prevent the contacts from welding for heavy-current relays

(6) Satisfactory life for the contacts at the operating environment

(7) Low contact resistance in low-current contacts

(8) Resistance to shock, explosive vapor, corrosion, radiation, etc.

Relay classification: The various relay designs, constructions, and applications are so numerous that many classifications are needed to categorize them. One method of classification is according to the basic form or construction, as follows:

(1) Hinged armature type

(2) Solenoid-operated type

(3) Rotary action

(4) Polarized

Each group can also be divided into subgroups according to their current rating, supply voltage, order of making and breaking, whether normally open or closed, sealed or unsealed, etc.

The hinged armature relay is one in which the armature is located on its pivots adjacent to the magnetic system. The relay contacts are either mounted on the armature or are actuated by the movement of the armature or an extension of it. A leaf or a spring is usually employed to hold the contacts in the "off" position when they are not energized.

The solenoid operated relay has the armature in the form of a core within a wound coil or solenoid. When the relay is energized, the core is driven to the center of the coil where the attraction is zero.

Solenoid actuation of relay contacts is used where relatively large movement of the contacts and considerable contact pressure are required. Similar to the hinged armature type, solenoid operated relays are sensitive to acceleration forces. Although the armature can be of a balanced type to reduce the acceleration effect, the additional weight and size necessary for proper balancing is rarely justified.

The construction of the rotary-action relay is similar to that of a motor. The armature is supported on a rotating shaft and is enclosed within a magnetic system. The armature may be the wound coil type while the field is a permanent magnet or vice versa; or both armature and field may be wound. When the relay is energized, the shaft is driven by a circumferential pull against the torsion of the return spring. The contacts are usually operated by a cam or lever on the shaft.

This rotary-action relay has inherent stability against acceleration forces. It is manufactured in a wide variety of sizes and shapes and has been used in mobile ground equipment, aircraft, and missiles.

Polarized relays are those in which the direction of the armature movement is dependent on the direction of the current flow in the field coil. The relay may be designed such that the contacts are closed only when the current flows in one direction: movement in the other direction is limited by a mechanical stop. The relay could also be designed such that different sets of contacts are closed according to the direction of the current flow.

Although polarized relays are suitable only for dc operation, they are very useful in paralleled dc generator systems. It is necessary, however, to accurately center the armature of the polarized relays since it is the most important factor affecting relay performance.

Contact materials: The voltage across the contacts, the operating frequency, and the type of load and environmental conditions vary within broad ranges for various applications. So the relay can operate successfully to satisfy a certain requirement, the choice of proper contact material becomes very important, particularly for relays designed to switch low current and low voltage circuits (1 mA and 100 mV or less) which are also referred to as dry circuits. For these circuits the most commonly used materials are gold or gold alloy, although various metals or alloys of the platinum group are also employed. Low level relays are generally considered to be much longer life devices than their more heavily loaded counterparts. A life of a few million to over a billion operations is typical, depending on the load and relay design.

In addition to the materials listed in table 11, other materials are being evaluated for relay contact applications. These include palladium, palladium silver, iridium-platinum and sintered silver graphite.

Contact pressures: For certain materials, the contact resistance depends primarily on the contact pressure; and the relation is consistent over many cycles. Fig. 35 shows a typical curve relating contact resistance and contact pressure for switch contacts with copper plane surfaces. It is seen that the contact resistance is inversely proportional to the contact pressure and when the pressure drops below a certain value, the contact resistance increases rapidly. Therefore, in order that the contacts give a fixed contact resistance, a certain minimum pressure is required. This requirement is one of the most important design considerations, since it directly affects relay performance.

High speed relays: No specific operating time has been established to qualify a relay as high speed. However, dc operated relays achieve speeds ranging to less than 1 msec. High speed is obtained through use of low moving mass, short travel, low eddy current, and often by use of polarized magnetic structures.

Operating time is also a function of the energized circuit. Overdrive (abnormally high coil voltage or current) accelerates relay operation, as will a low impedance energizing source. High energy pulses of short duration are used for high-speed polarized latching relays. It should be pointed out that coil

TABLE II

MATERIALS AND APPLICATIONS, ADVANTAGES AND DISADVANTAGES

Material	Application	Advantages	Disadvantages
Silver	General application	Relatively inexpensive; high thermal and electrical conductivity; requires low contact pressure	Produces silver sulphide coating (the resistance becomes important at low voltages)
Rhodium plated silver	Similar to that of silver (if arcing does not break down the plated film)	Same as silver; prevents formation of sulphide film	Expensive
Platinum	Low current (less than .5 A) low voltage; super sensitive relays	Requires very low contact pressure; very good resistance to corrosion	Very expensive
Gold	Dry circuits	Requires low contact pressure; very good resistance to corrosion	Expensive
Tungsten	Low current or nonsensitive low voltage circuits	High melting point; relatively low cost; not effected by arcing, inductive currents, or corrosion	Requires high contact pressure
Sintered nickel-silver	Heavy inductive current circuits	Requires moderate contact pressure; good electrical conductivity; less sensitive to sulphur	Contact resistance may increase with use

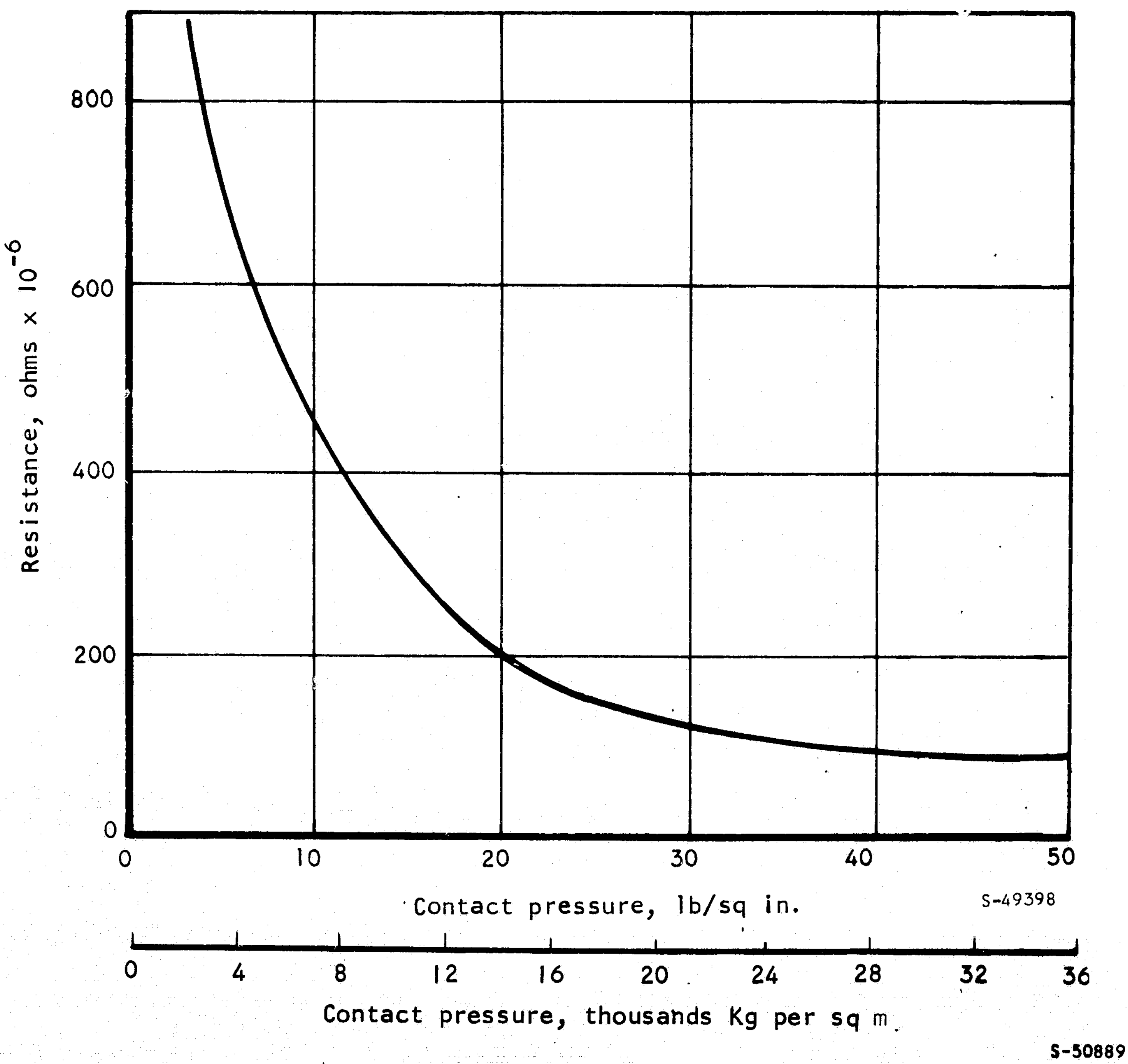


Figure 35. Contact Resistance vs Contact Pressure for Switch Contacts with Copper Plane Surfaces

overdriving or abnormal acceleration of the relay operation may result in increase contact bounce. If operated on fast duty cycles, high levels or energization can tend to reduce dropout speeds.

High-speed ac relays are also available. Regardless of the point on the sine wave at which the coil is energized, operation within less than half-cycle time duration is possible.

Typical specification for a general-purpose aircraft relay (10 A, 4PDT): Typical specifications are listed below.

Min. operating cycles	100,000 (except lamp and dc motor load--50,000)
Max. operating time (dc)	.010 sec
Max. operating time (ac)	.015 sec
Max. release time (dc)	.010 sec
Max. release time (ac)	.015 sec
Max. duration of contact bounce	.003 sec N.O. .005 sec N.C.
Rated duty	continuous
Operating temperature	-70° to $+125^{\circ}C$
Dielectric strength:	
at sea level--coil	1250 V rms
contacts	1250 V rms
at altitude--coil and contacts	350 V rms
Vibration resistance	.08 D.A.--5-10 Hz .06 D.A.--10-80 Hz ±20 G.--80-2000 Hz
Max. weight	.20 lb
Internal volume	≈ 1-in. cube (16.4 cm^3)
Unless otherwise specified, tolerances: Decimals	±.010
Applicable specifications	MIL-R-6106-D MS-27255-2 MS-27255-1 MS-27254-2 MS-27254-1

Contact rating (table 12): Contact rating is the maximum current for a given type of load, i.e., the voltage, current, frequency and nature of impedance which the relay contacts will make, carry and brake for its rated life or for the number of open and closed operations. Contacts must be derated for inductive loads. (See fig. 36.) For example, a 10-A contact handling load of 60 percent pf must be derated by .86 or to 8.6 A for maximum effective life.

TABLE 12

CONTACT RATING

Contact rated load (Ampere per Pole)

Type of Load	28 Vdc	115 V 400 cycles ac	115/200 V 400 cycles ac
Resistive	10	10	10
Inductive	10	10	10
Motor	5	5	5
Lamp	3	3	3
Minimum	Minimum current per specification	Minimum current per specification	Minimum current per specification

Coil		Nominal	Maximum	Pickup	Dropout
dc	Voltage	28	29	18	7, +0, -5.5
	Current	----	.20	----	----
ac	Voltage	115	120	90	30, +0, -25
	Current	----	.04	----	----

Circuit breakers.-- Two basic types of circuit breakers are used in aircraft: thermal breakers and magnetic breakers. Thermal tripping is affected by a bi-metallic strip heated either directly by the current it carries or indirectly by a heating element in series with the load. The thermal circuit breaker is affected by the ambient temperature, although this is not so much of a disadvantage as it first appears because some kind of temperature compensation can be provided if necessary. One of the advantages of this type of circuit breaker is that it has an inherent time delay. Magnetic tripping is actuated by a solenoid which is sensitive to excessive current or voltage. This type of circuit breaker is unaffected by the ambient temperature; it has almost instantaneous operation; hence, any required time delay has to be achieved by using a mechanical device such as a dash-pot, air-brake, etc.

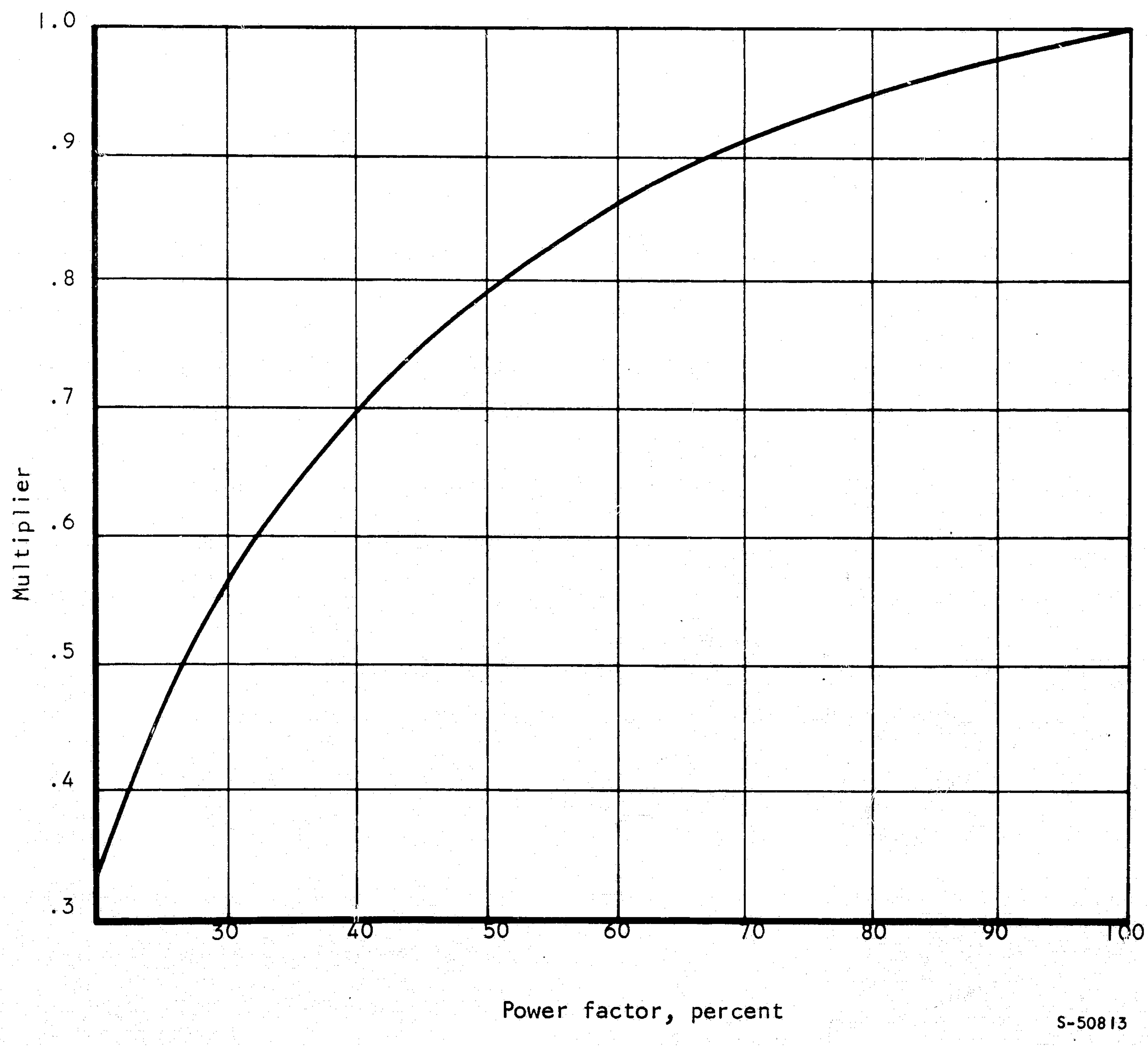


Figure 36. Contact Ratings for Power Factors other than Unity

In selecting a circuit breaker for a load circuit, several factors must be considered before a proper rating can be determined to match the capacity of the wire. These factors are:

(1) The steady-state conditions of the load versus the steady-state capacity of the wire

(2) Startup and transient overloads together with their durations

(3) The ambient operating temperature and the means of heat dissipation

(4) The environmental conditions under which the circuit breaker must operate

(5) The possible voltage and frequency variation of the circuit

(6) The allowable voltage drop

(7) The required trip-time delay

(8) Coordination with other breakers connected in series on the same line

For most applications, in order to avoid nuisance tripping, the circuit breakers are required to pass the startup and transient overloads (which are of the short-duration type) as well as a continuous overload slightly higher than normal. Consequently their actual trip characteristics are higher than their rated current capacity. The thermal circuit breakers are designed with both a maximum and a minimum ultimate trip. The maximum ultimate trip is the percent overload under which the breaker must trip within one hour, while the minimum ultimate trip is the percent overload under which the same breaker must remain closed for at least one hour before tripping. Although 138- and 115-percent overloads are the typical maximum and minimum ultimate trip respectively, maximum limits as high as 180 percent and minimum limits as low as 100 percent may be required for specialized applications.

To ensure that the circuit breaker can provide optimum protection under faulty conditions while preventing nuisance tripping under transient conditions the circuit breaker must be matched to both the short duration overload characteristics and the wire capacity. Fig. 37 shows qualitatively the characteristics of multi-conductor wire based on wires in bundles. Since each wire has a maximum allowable operating temperature, the current carrying capacity of the wire is a function of time.

Thermal breakers.--Fuses and thermal breakers respond according to the function $I^2 t$ where I is current, t is time, and the resistance is assumed constant. Magnetic breakers operate as a function of current (I) only, the coil turns being constant. Both thermal and magnetic breakers require an appreciable time to operate. Magnetic types are by far the faster; when there is a dead short circuit the mechanical mechanism of a fast magnetic breaker will respond in as low as 3 or 4 msec. Because the primary concern in the protection of aircraft distribution systems is protecting the wire from excessive overheating

and the eventual burning which follows a thermal-time characteristic curve, a protective device with a similar trip-time characteristic is desired. For this reason the thermal breaker is widely used in aircraft while the magnetic-breaker has only rare application where special protection of some load equipment is desired.

(1) Temperature effects--Since tripping of the thermal circuit breaker is initiated by the temperature rise in the heating element, any environmental condition which affects the temperature rise would affect the performance of the circuit breaker, including externally induced heat, internal heat dissipation, etc.

When the circuit breaker is subjected to extremes of temperature environment, serious derating of the circuit occurs. In general, degradation by temperature can occur from the following three basic sources.

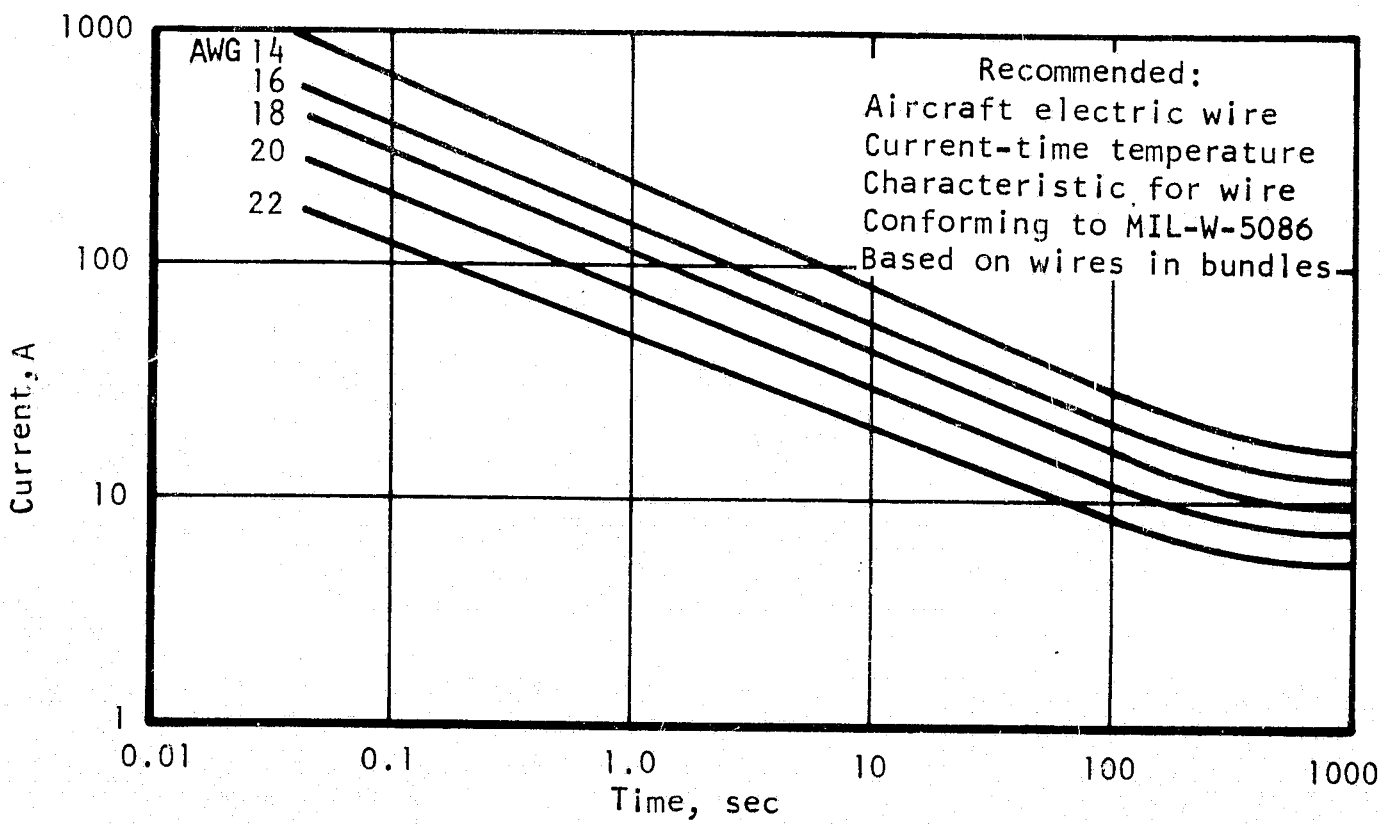


Figure 37. Current Amperes vs Time

(a) Temperature variation--If the circuit breaker is installed in such an environment where the temperature varies in a broad range, the trip capacity of the breaker will be seriously affected. A circuit breaker rated at 115/138 percent at a nominal temperature of 25°C would have a significantly higher trip capacity at low temperature range and a very low trip capacity (in some cases, even below the normal rating) at high temperature. The amount of change in trip capacity is directly related to the temperature deviation. When the trip capacity is raised due to low ambient temperature, the circuit breaker would not provide proper protection to both the equipment and the wire. On the other hand, when the trip capacity is lowered due to high ambient temperature, premature tripping would occur.

(b) Preloading--When the circuit breaker carries a load at or near full load for a relatively long period, it is derated due to the sustained heating. If the breaker is subjected to a short duration transient load, premature tripping may occur. Fig. 38 illustrates the effect of preloading on trip times of a bi-metallic thermal breaker. The dotted line shows the load vs trip time with no preload, while the solid curve is for 100-percent preload. Since the effect is considerably linear, tripping characteristic curves with partial preload can be determined by interpolation between the curves.

It should be mentioned that magnetic circuit breakers are also subjected to degradation due to preloading, although they are not actuated by temperature change. The effect, however, is nonlinear and more change in trip time is observed as full preload is approached.

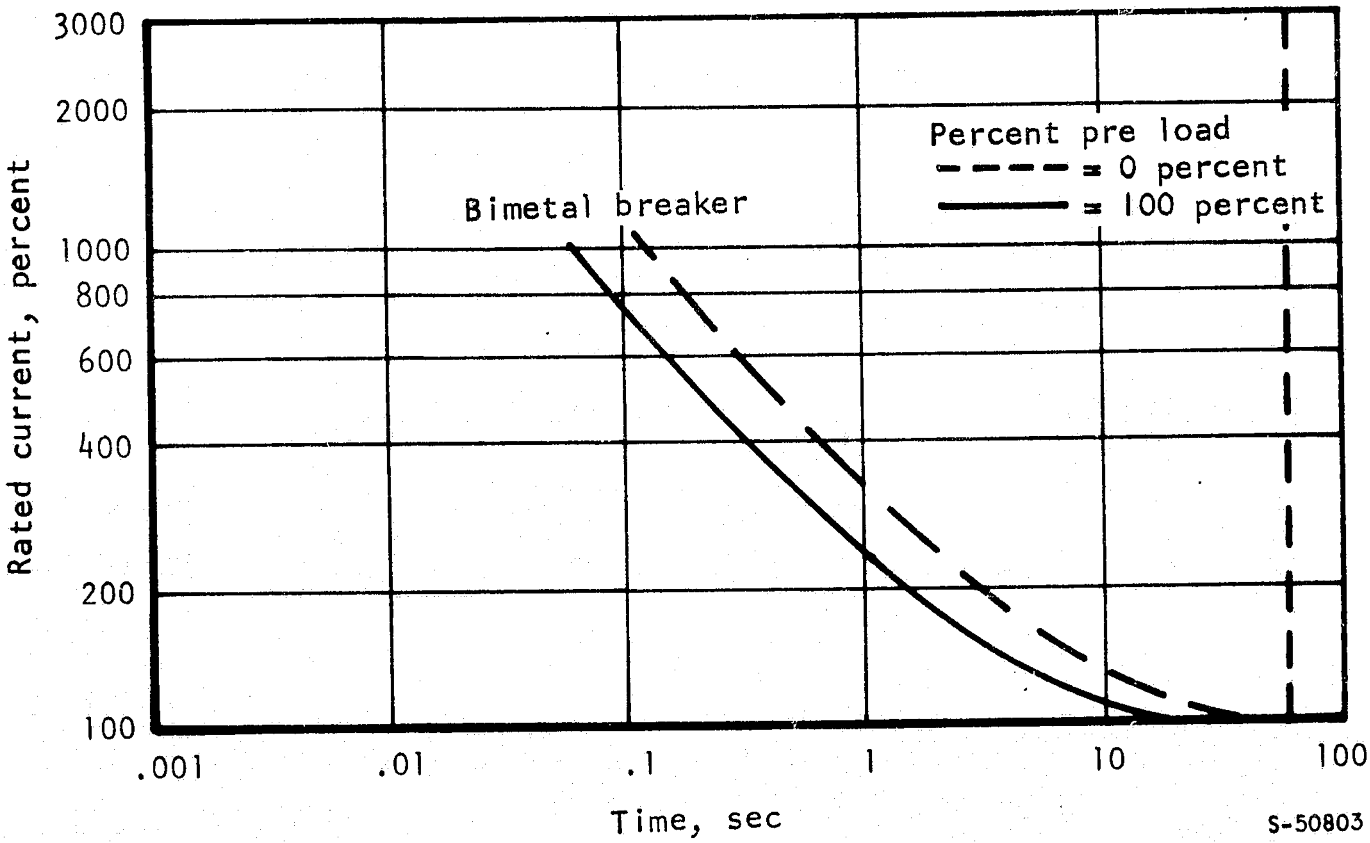


Figure 38. Percent of Rate Current vs Time

(c) Heat dissipation--The heat dissipation capability of the breaker is closely related to the size and material of the case, wire terminals, mounting, and the surrounding atmosphere. These factors which affect the temperature rise are taken into consideration by the designer.

When a great number of breakers are mounted close together on a single panel, adequate heat transfer may be hampered and overheating results. As a consequence the circuit breakers have to be derated. Proper ventilation, therefore, should be provided to improve heat dissipation especially when the air surrounding the breaker is rather confined.

Altitude has a direct effect on the heat dissipation capability of the circuit breaker, simply because at high altitude, the air density decreases and therefore the ability of transferring heat by convection is reduced. A circuit breaker having a maximum ultimate trip of 138 percent at sea level would be derated to approximately 130 percent at 50,000 ft (15,200 m) and 120 percent at 100,000 ft (30,500 m).

It should be noted that in some applications, the wiring of a certain load may pass through a heat source such as the engine zone, the aircraft heating system, etc. while the circuit breaker is not exposed to it. Under this condition, the wire is subjected to a preload, and a slight overload might cause the wire to fail before the breaker is actuated. The maximum ultimate trip of the breaker, therefore, must be lowered in order to protect the wire properly; it may be necessary to re-route the wire if the breaker trip limit cannot be reduced due to the peculiar load characteristics.

(2) Interrupting capacity--Interrupting capacity of the breaker is the maximum current which the breaker is able to interrupt. This interrupting capacity is a function of many factors, among which the most significant ones are the size and geometry of the breaker, the line voltage, and the current rating. In general high interrupting capacity is desirable; this requirement, however, is incompatible with the requirements of small size, light weight, etc. Low-rating breakers have smaller interrupting capacity than high-rating breakers because of the smaller physical size.

It had been a strict rule, in the design of aircraft electrical circuit protection, that a circuit breaker should not be used in a circuit in which the short circuit current exceeds the interrupting capacity of the breaker. This is to prevent the breaker from being destroyed by the short circuit current and thereby leaving the circuit either unprotected or disconnected. Recently, however, a modification of this design philosophy has come under consideration.

Many nonessential loads are not critical to flight safety of the aircraft. If a circuit breaker having a lower interrupting capacity than the short circuit current is utilized in a nonessential circuit; and if the breaker can guarantee an open circuit after the destruction of the breaker, the most serious situation that can be encountered is the loss of the circuit after a destructive fault. Experience, however, has revealed that this type of fault during a flight is extremely rare. Therefore, if this can be tolerated by some of the nonessential circuits, many advantages can be realized. The most obvious ones would be the savings in volume and weight since smaller breakers are selected.

Today, circuit breakers which guarantee an open circuit following the destruction of the breaker are available. This type of breaker is referred to as "fail safe."

(3) Voltage drop: Some utilization equipment are very sensitive to the terminal voltage. In calculating the total voltage drop of the circuit, therefore, the circuit breaker in series with the load must be taken into consideration if

the expected performance of the load equipment is to be realized. Bimetallic thermal breakers have low voltage drops (.5 V) and usually do not create any problems. However, the voltage drop should not be ignored in making a selection.

Thermal breakers with rating below 5 A are usually of the indirectly-heated type; i.e., the trip device is heated by a wire-wound element. This type design is not desirable since it introduces an excessive voltage drop in the circuit.

(4) Types of thermal breaker--There are two basic types of thermal breakers:

Switch-type breaker--These breakers are available in a variety of sizes and actions. Some of them are miniaturized and a single unit can frequently replace the toggle switch and the fuse.

Manual reset breaker--This breaker is most widely used in aircraft. The breaker has one button on its face. Manual reset is done by pushing the button which will come out when it is tripped on overload. Most models can be tripped manually by pulling out the button.

The current rating of these breakers for aircraft use ranges from less than 1 A up to 50 A for the ac system. A typical miniaturized breaker in this range weighs about 1.5 oz.

For general safety all breakers used on aircraft must be of the trip-free type. That means that the breaker is designed so that it will not maintain the circuit closed when carrying overload current regardless of the restraint placed on the actuator.

There are multi-pole circuit breakers constructed by combining several single-pole circuit breakers. It is designed so that whenever one pole is tripped the remaining pole will trip simultaneously. A three-pole breaker can be used for three-phase protection while a two-pole breaker can be used for a two-wire circuit or can be used as an interlock between two circuits serving the same equipment.

(5) Tripping time--Curves showing trip time-versus-load are usually provided for each circuit breaker to serve as a guide for application purposes. Figure 39 shows a typical trip time characteristic at 25°C. A band rather than curve displays the manufacturer's or specification tolerances for a certain breaker rating. There are three regions in the diagram. In region 1, below the shaded area, the circuit breaker should not be tripped; in region 2, the shaded area, tripping may occur; in region 3, above the shaded area, the circuit breaker should definitely be tripped.

The temperature effect on trip time characteristics is shown in fig. 40. It is understood that temperature above 25°C tends to move the curve downwards while ambient temperature below 25°C tends to move the curve upwards. If the ambient temperature varies between -54° to +71°C the shaded area of fig. 39 will be enlarged as shown in fig. 40. Temperature effects can be eliminated or reduced by employing a temperature compensating unit to control the thermal element.

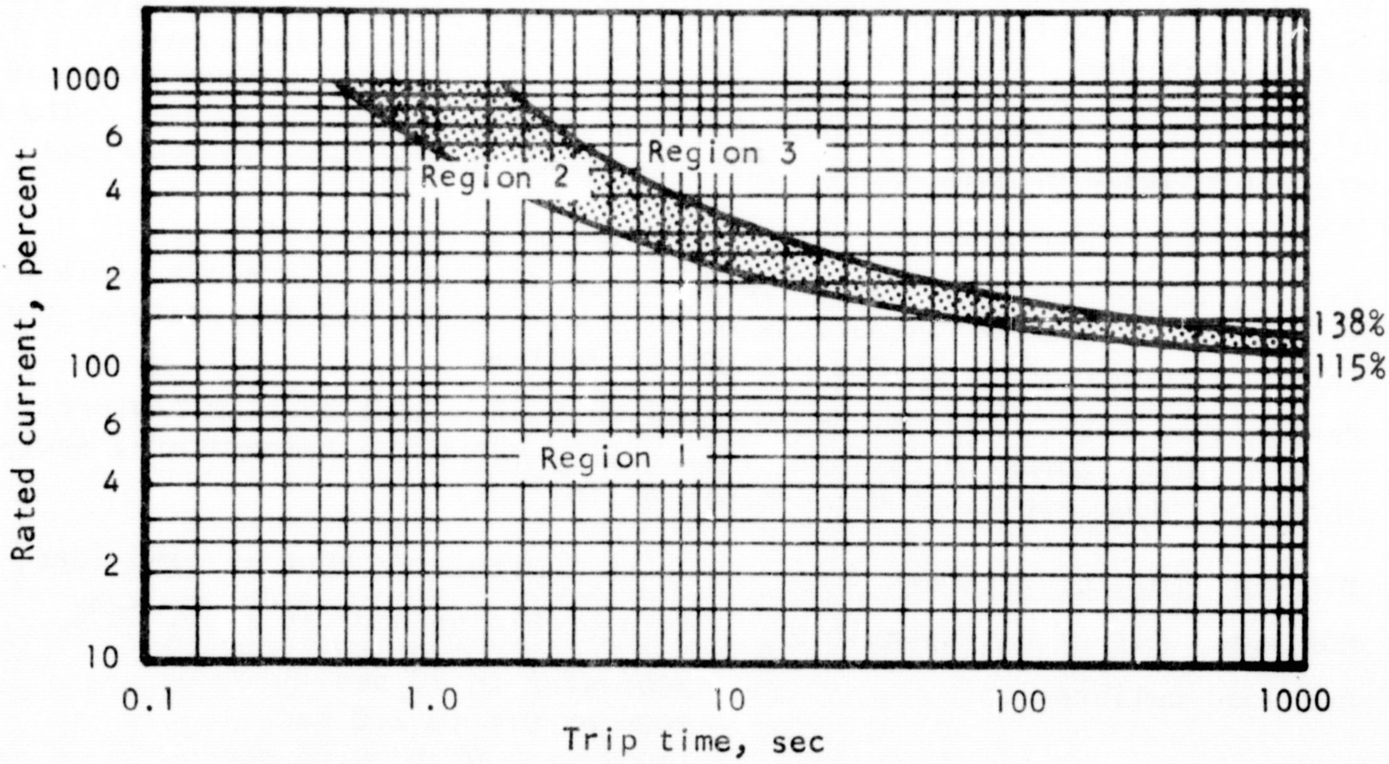
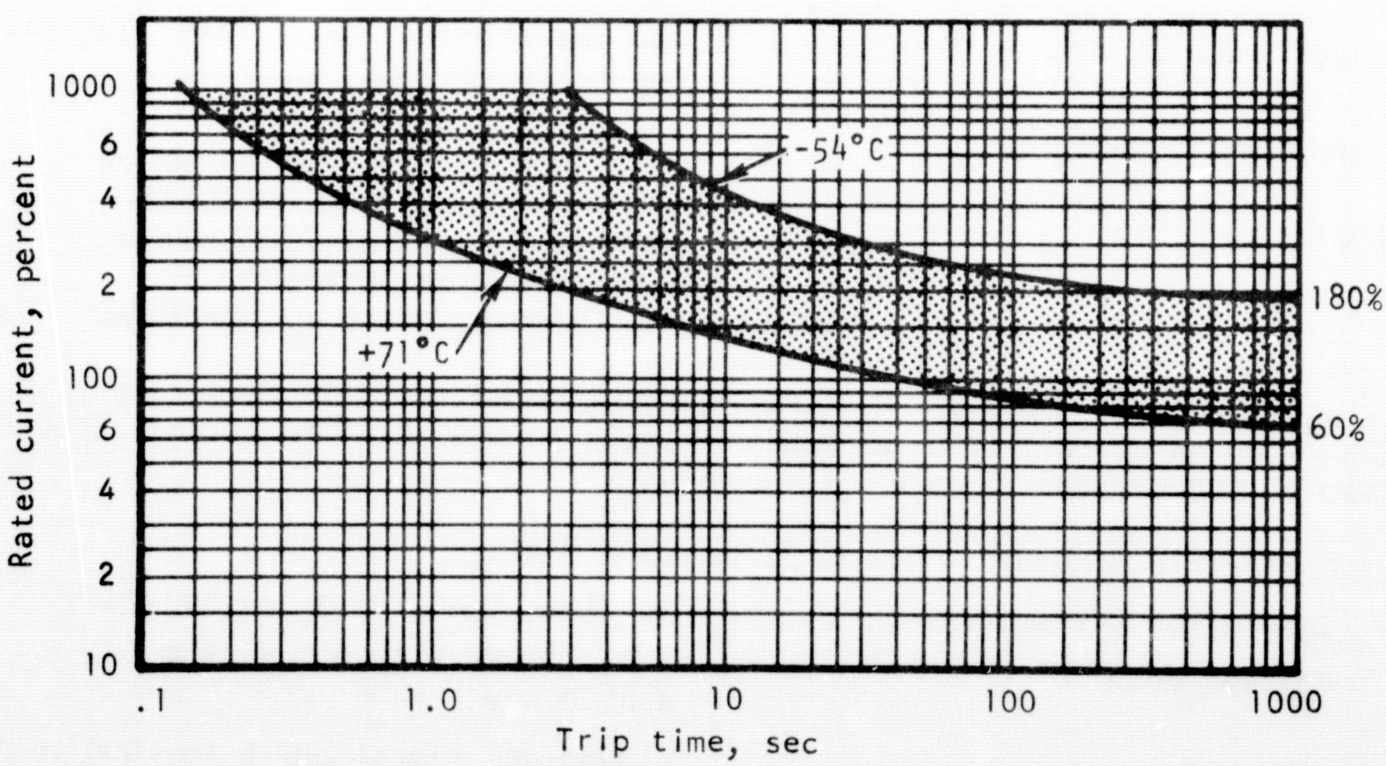


Figure 39. Tripping Time Characteristic of Circuit Breaker



S-50835

Figure 40. Temperature Effect on Tripping Time of Circuit Breaker

(6) Circuit breaker coordination--A basic problem which must be considered in circuit breaker application is coordination. When circuit breakers are series connected with higher rating breakers or fuses in a distribution system, it is desirable that only the breaker with the lower rating trip under any overload or fault condition. Otherwise whole sections in a distribution system would be de-energized instead of just the circuit branch where the fault or overload occurred.

(7) Performance data--Typical performance data of a late model miniature circuit breaker that comes in ratings of 1 to 4 A, meets the temperature, shock, and vibration requirements of MIL-C-5809D as follows:

Performance data: Typical performance data of a late model miniature circuit breaker that comes in ratings of 1 to 4 A, meets the temperature, shock, and vibration requirements of MIL-C-5809D as follows:

Minimum limit of ultimate trip	No trip within an hour at 115% load, 25°C
Maximum limit of ultimate trip	Trip within an hour at 145%, 25°C
Overload calibration	200% at 2 to 20 sec 500% at 0.1 to 2.0 sec 1000% at 0.02 to 0.05 sec
Ambient effect on calibration	At -54°C, 135% load, no trip in one hour; 180% load will trip within one hour. At +71°C, 90% load, no trip in one hour; 130% load will trip within one hour.
Overload cycling	100 operations at 200% rated current and rated voltage
Rupture capacity	1 A, 6000 A at 30 Vdc 3500 A at 120 V, 400 Hz ac 2 and 3 A, 6000 A at 30 Vdc 1500 A at 120 V, 400-Hz ac 4 A, 3000 A at 30 Vdc 1500 A at 120 V, 400-Hz ac

Where 2-, 3-, and 4-A units are rated below MIL-C-5809D requirements, units are designated as fail safe at rupture levels to 6000 A at 30 Vdc and 3500 A at 120 V, 400 Hz ac.

Dielectric strength	1500 V min.
Insulation resistance	Not less than 100 MΩ at 500 Vdc
Voltage drop	1 A, 1.1 V; 2 A, 0.7 V; 3 A, 0.5 V; 4 A, 0.4 V
Vibration	Sinusoidal vibration: 5 to 1000 to 5 Hz at 10 g

This breaker weighs less than 30 grams and is approximately 1.5-in. (3.8 cm) long.

Magnetic breakers:--The magnetically actuated breaker contains a solenoid that generates a magnetic flux proportional to the load current. There are no thermal elements and practically no heat is generated within the breaker. As a result, the breaker is free of adverse ambient-temperature effects. Its nominal current rating and calibrated trip-points are fixed and remain constant regardless of ambient temperature fluctuations. Derating is never necessary. The breaker will carry 100 percent of rated load and will trip as specified at any temperature within its tested operating range. Since some time delay is desired for most applications, magnetic breakers incorporate a mechanical time delay such as an iron slug moving in oil. Although the trip-time characteristic of the breaker itself is unaffected by ambient temperature, the delay device is temperature dependent because the oil viscosity changes with temperature.

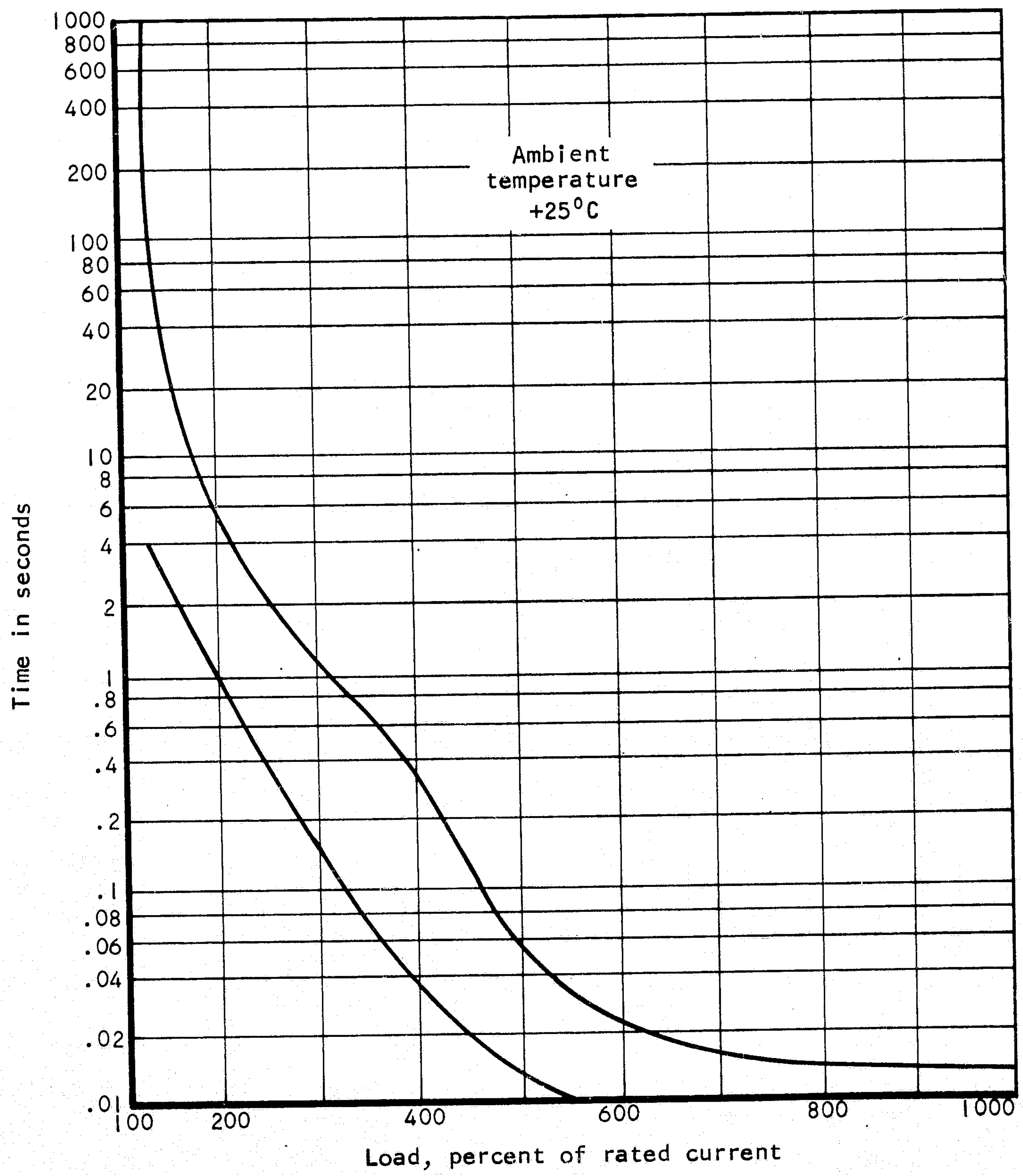
Higher ambient temperatures shorten the time-delay period. This is desirable because equipment overload tolerance is reduced under higher temperature conditions. Conversely, lower ambients lengthen the time-delay period, providing extra time in which to start cold equipment with safety. The effect of temperature on a magnetic breaker is illustrated in fig. 41 (from ref. 10). The current of trip remains unchanged. The trip time of a certain type of magnetic breaker is 60 msec at 25°C, 90 msec at -40°C, and 35 msec at 100°C when there is 400 percent of rated current. The 200-percent trip is 1.5 sec at 25°C, swinging from 12 sec at -40°C to 1 sec at 100°C. The faster time to trip, with a constant trip current, appears to be advantageous. A magentic breaker can be reset immediately after tripping, although the delay mechanism does not immediately reset. If the fault is still present this will reduce the time to trip. This is usually not true, however, with respect to thermal breakers since the heating element must cool down before it will reset.

The magnetic breaker shown in fig. 42 (from ref. 10) is essentially a toggle switch comprising a handle connected to a contact bar by a collapsible link. The contact bar opens and closes an electrical circuit as the handle is moved to the ON or OFF position. When the link collapses, the contacts of the unit open, breaking the electrical circuit.

The magnetic circuit within the unit consists of the frame 1, armature 2, delay core 3, and pole piece 12. The electrical circuit consists of terminal 4, coil 5, contact bar 6, contact 7, contact 8, and terminal 9.

As long as the current flowing through the unit remains below 100 percent of the rated current of the unit, the mechanism will not trip and the contacts will remain closed as shown in fig. 42.

If the current is increased to between 100 and 125 percent of the rated current of the unit, the magnetic flux generated in coil 5 is sufficient to move the delay core 3 against spring 11 to a position where it comes to rest against pole piece 12 as shown in fig. 42b. The movement of this core against the pole piece changes the forces of the magnetic circuit, described above, enough to cause the armature 2 to move from its normal position shown in fig. 42a to the position shown in fig. 42b. As the armature moves it trips near pin 13 which in turn triggers the collalsible link of the mechanism thus opening the contacts.



(a) 25°C ambient temperature

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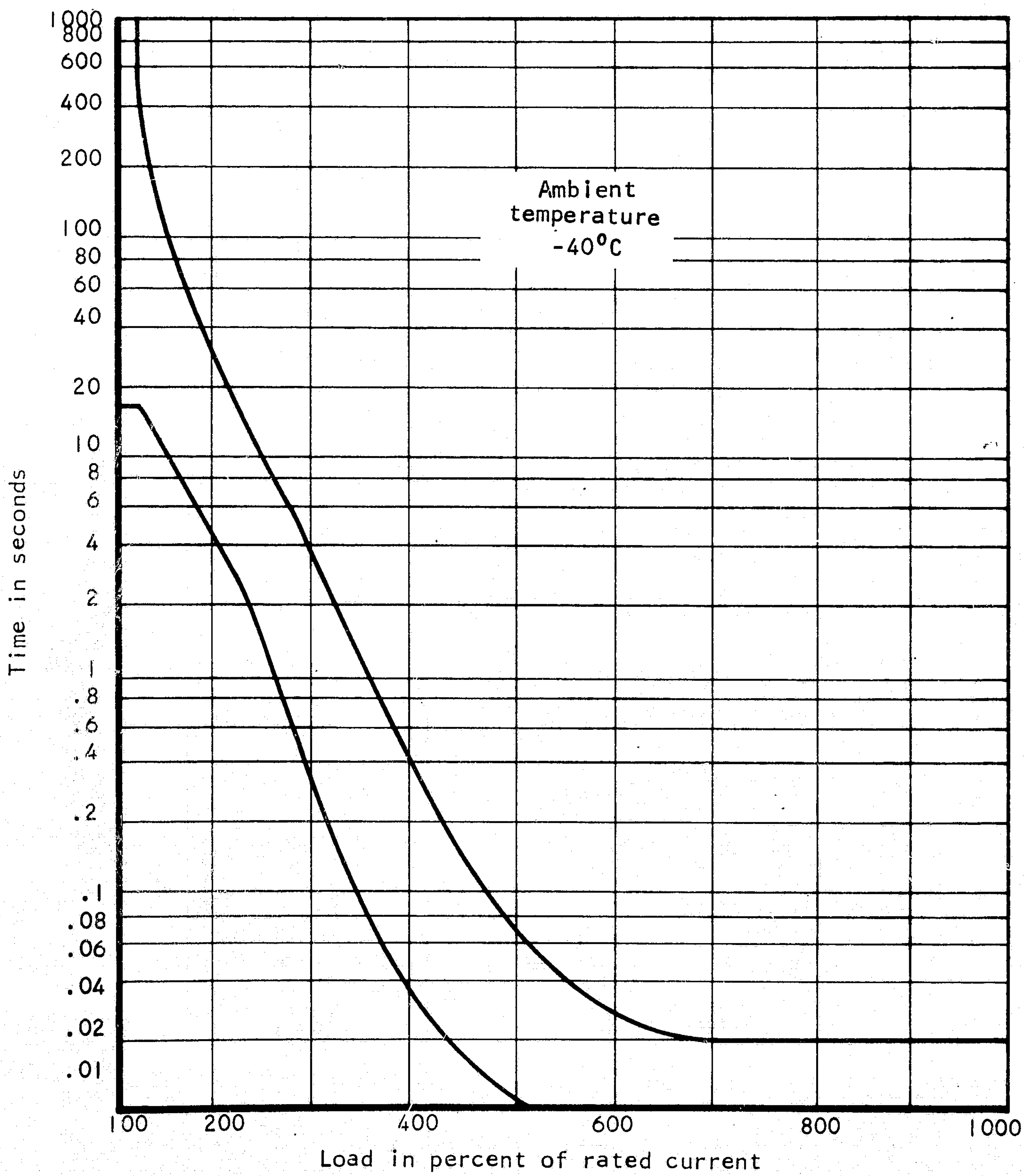
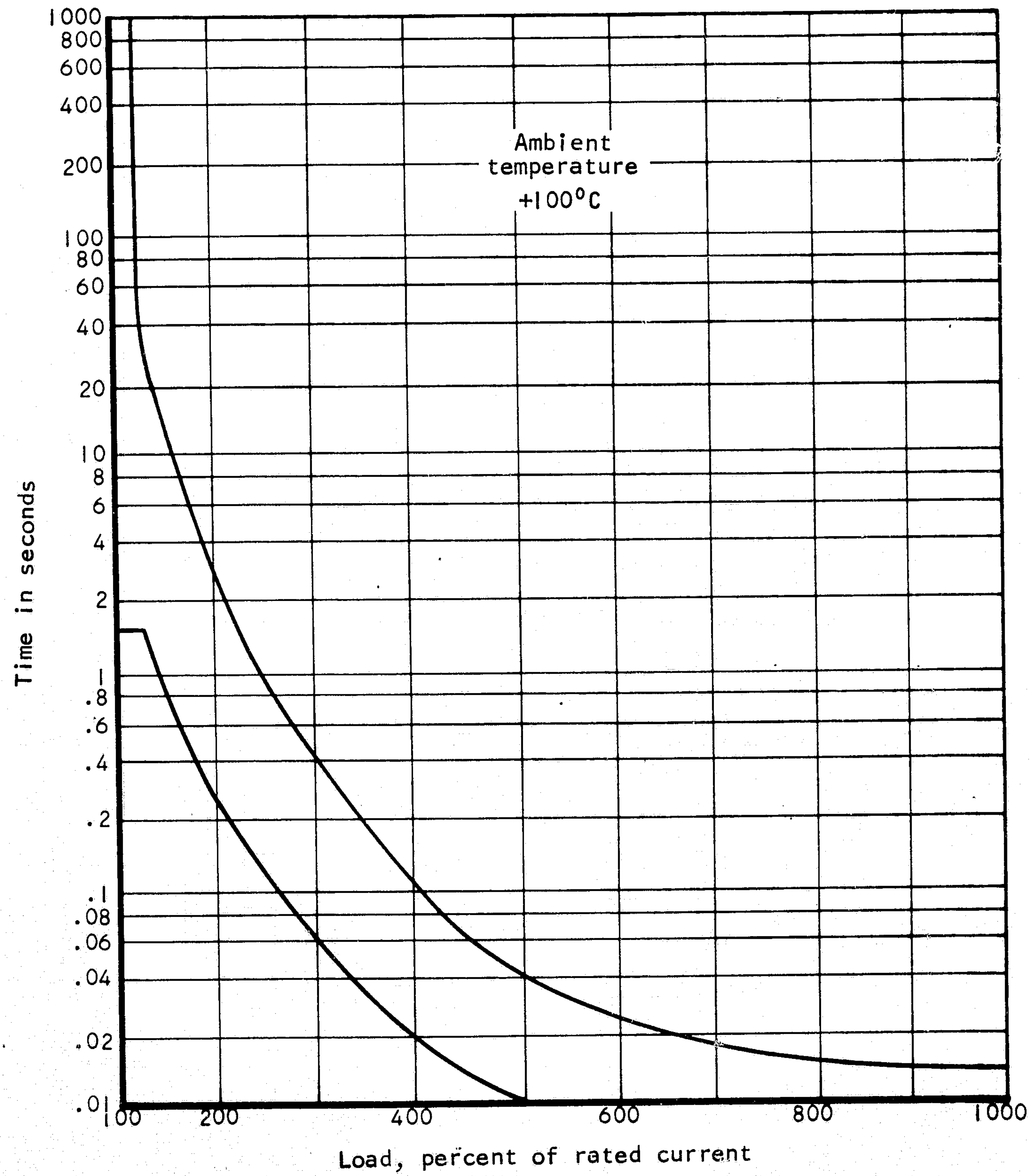
Figure 41. Effect of Temperature on Magnetic Breaker Trip-Time (Continue to next two pages)



(b) -40°C ambient temperature

S-50812

Figure 41. Continued



(c) +100°C ambient temperature

S-50810

Figure 41. Concluded

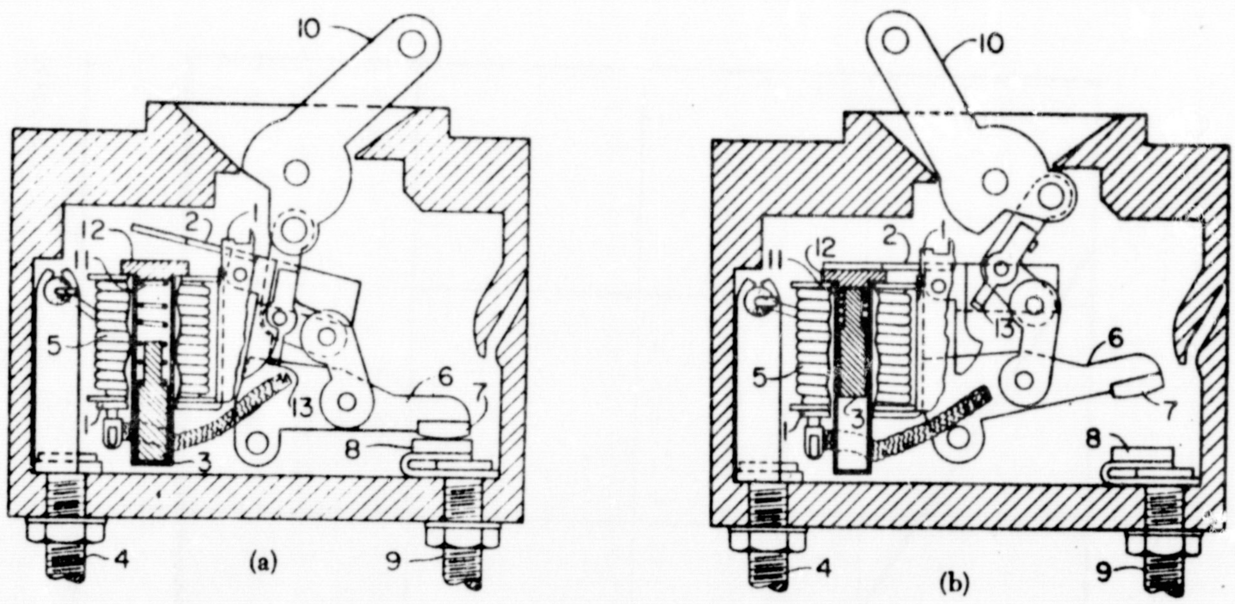


Figure 42. Operating Mechanism of an Airpax Magnetic Circuit Breaker (ref. 10)

The delay tube is filled with a silicon fluid which controls the speed at which the delay core moves. Different delay curves can be obtained by using fluids of different viscosities. When surges of 600 percent and more occur in the electrical circuit, however, the magnitude of the flux produced in the magnetic circuit is sufficient to trip the unit without the delay core changing position. Therefore, the delay time at currents over 600 percent is very short.

The speed of a magnetic breaker can and does give rise to nuisance tripping on high-speed transients (essentially in an rms response). If a transient of sufficient energy content arrives, the breaker responds in the "instant trip" mode as it does for the short circuit case. Irrespective of core position, this mode is sufficient to pull in the armature. To partly compensate for this undesirable effect in the magnetic breaker, breakers were made available with inertial integration of short time pulses. The integrator (inertia wheel in the armature trip mechanism) affects only the armature; it does not control the longer time delays to any appreciable extent. In effect, the integrator makes the breaker responding to the RMS value of current averaged from a time period containing many cycles.

Magnetic breakers are available in ratings from 50 mA to 20 A. For voltages up to 250 Vac, 400 Hz, and 50 Vdc. However, they are slightly heavier and also somewhat higher in price than thermal breakers.

Transformer-rectifier units and inverters.-- Typical weight vs rating for present aircraft transformer-rectifier (TR) units is shown in fig. 43. Silicon diodes are used exclusively in rectifier units.

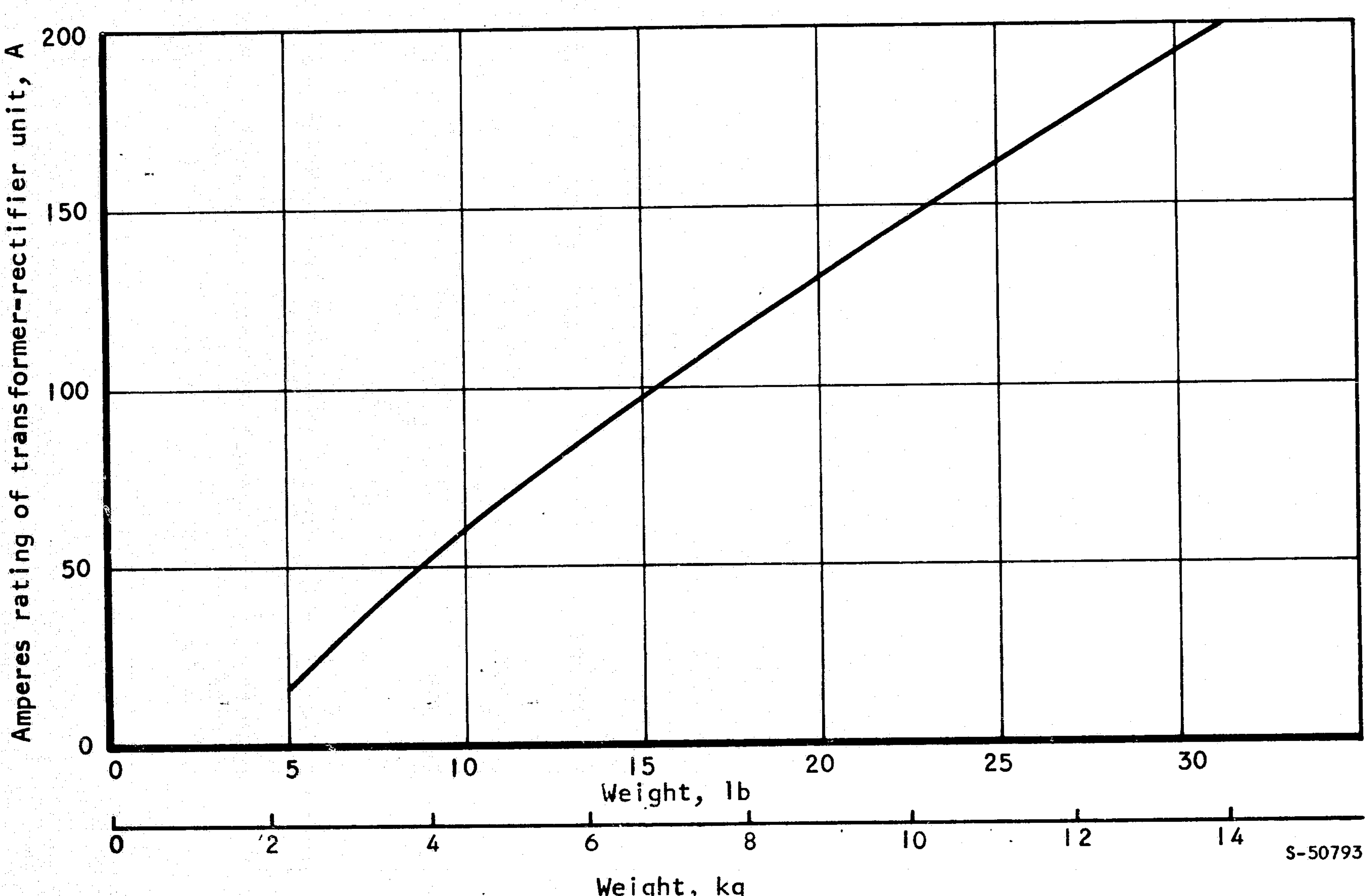


Figure 43. Weight of T-R unit (28 vdc)

When rectifier units are operated in parallel, a derating factor must be used to allow for unequal load sharing. A practical formula for the derating factor is

$$K = \frac{(1 - LD)N + LD}{N}$$

where N = number of units operating in parallel

LD = allowance in difference of load division (a normal value for this quantity is .2)

Static inverters installed in existing commercial aircraft are small transistor units (500 VA or less), converting 28 Vdc to 115-V, 400 Hz, single-phase ac. A typical 500-VA unit weighs slightly less than 20 lb (9 kg) and has an efficiency of about 75 percent. Inverters larger than 500 VA in existing aircraft are usually rotary type.

Typical weight of present inverters is shown in fig. 44.

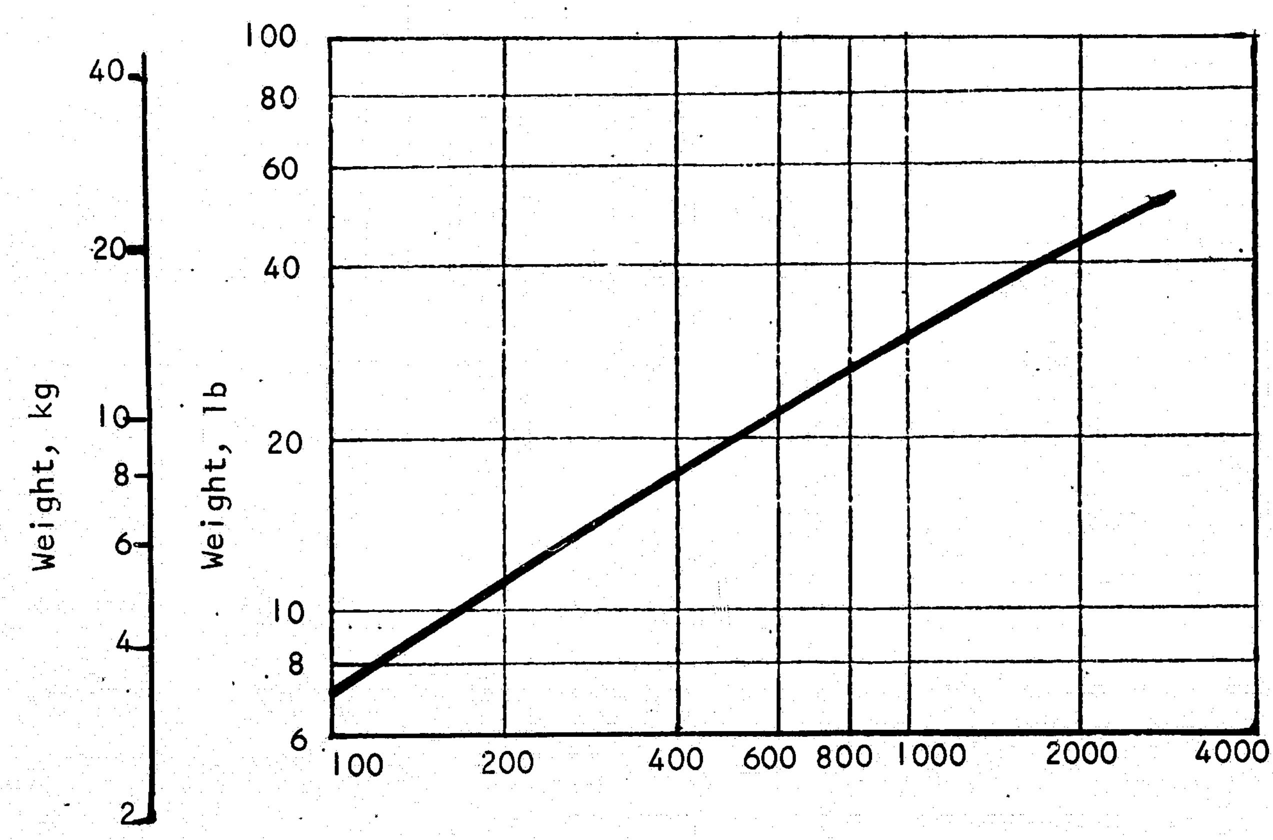


Figure 44. Typical Weight of Aircraft Inverter Single or Three Phase 28 Vdc to 115 V or 115/200 Vac

3

Connectors.--Service experience has shown connectors to be one of the more vulnerable components causing malfunction of the electrical system. Incidents of open circuits, intermittent contacts, and high circuit resistance due to corrosion are reported repeatedly. The need for reliability improvements in circuit interconnections is more eminent than for any other component in the system. The ideal case, as far as reliability is concerned, is to have continuous conductors throughout the entire circuit without using connectors at all. However this usually is not the case. Interconnections often are added to facilitate assembly, maintenance, future modification, inspection, and troubleshooting.

At circuit design levels the principal considerations of interconnection are related to the selection, rating, and derating of individual conductors. As the design of electrical/electronic equipment nears the stage of packaging and installation all the individual conductors begin to be placed in orderly channels, multi-conductor cables, or other mechanical combining modes. More often the stage at which this channeling is considered is far too late to avoid expenditures of time and money. Mechanical restrictions have already been built into the packaging which minimize the number and size of cables and bundles that may be incorporated. Worse yet, the packaged equipment is often perfect in all respects except that no room has been left for adequate wiring.

The first phase in the design of any cabling or multiconductor interconnection mode is at the block diagram level, where planning of ultimate requirements start. As the equipment progresses through design and other refining steps to mock-up and manufacture, the interconnection system is developed along with it.

Interconnection requirements.--The following list contains a summary of various items a circuit designer considers when specifying interconnections. The list gives an idea of present practice--items do not appear in order of importance, nor do all of them apply at every stage of the design.

(1) Classes

Power

Signal

RF

Shielding

Thermocouple

Estimated resistance

Voltage drop for long lines

Environment free (classify each requirement)

Nonenvironment free

High voltage

Hermetic

(2) Groups

In conduit

Cabled

Harnesses

Single wire

(3) Growth factor

(4) Utilization factor

(5) Number of connector contacts

(6) Types of contacts

Open entry

Closed entry

Contact material needed for conductivity rating and environmental considerations

(7) Contact size (all wire gages)

(8) Connector sizes

Microminiature

Subminiature

Miniature

Standard

Heavy duty

(9) Current rating

(10) Voltage rating

(11) Shells

(12) Method of installation

Pendant, both halves

Pendant, one half

Rack and panel rectangular

Rack and panel cylindrical

Chassis mount board

Chassis mount tape

Standard pendant, rectangular

Standard pendant, cylindrical

(13) Couplings

Quick disconnect, no lock

Quick disconnect, lock

Push-pull

Pull-pull

Bayonet

Thread assist

Guide pin and lead screw

(14) Shell finishes

(15) Insert materials

(16) As per environment

(17) Polarization

Contact rating.--The electrical rating of contacts is a very important consideration in the selection of a connector during the circuit design stage. Contacts are designed to carry about the same amount of current as the corresponding wire size. And the wire entry portion is normally large enough to accept this wire size. The most frequently used contacts are the cylindrical pin- and socket-type made of copper alloy. The superior conductivity of copper permits smaller diameter pins and sockets than other metals. Table 13 (from ref. 11) lists various metals used for making contacts. Many of them are impractical for all but a few specialized uses because of their high relative resistance.

Cylindrical contacts, the most common, generally will not accept a wire larger than the gage size of the contact. However special types of contacts are available into which larger wires can be attached. These special connectors are useful where the resistance of smaller wire would be too great for the proper performance of the circuit, or where the smaller wire would not be physically able to take the abuse expected. Individual contacts and wires are capable of carrying considerable amounts of current when operating in free air. However this is an ideal situation. Most circuit designs require wires to be placed in close proximity to each other or often they are grouped together in a harness. When assembled tightly together they interfere with each other's ability to reject generated heat. Under these conditions their current carrying capacity is derated. Derating tables are available for comparing wires in budnles and single wires.

Connector criteria checklist.--After a list of connector requirements was formulated by component manufacturers and users throughout the aircraft industry, the list of criteria presented below was established. This list is presently used by designers to obtain design goals and performance conditions for developing miniature-environmental connectors.

Criteria No.	
1	Quick disconnect?
2	Positive lock without safety wiring?
3	Connector seal before lock?
4	Visually inspectable for correct assembly and installation?
5	Moisture seal?
6	Vibration dampener?
7	Corrosion resistant?
8	Operation to 250°F?
9	Unaffected by altitude pressure variations?
10	Wire comb?
11	Wire support?
12	Continuous dielectric separation--no voids?
13	Closed entry contacts to accommodate AN wires 22 through 18?
14	No wet process involved?
15	Good serviceability?

TABLE 13

CONNECTOR MATERIALS (ref. 11)

Materials	Relative resistance	Temperature coeff. of resistivity (part per deg. C)	Specific gravity	Coefficient of thermal conductivity ($\times 10^{-4}$ per deg. C)	Coefficient of thermal expansion ($\times 10^{-6}$ per deg. C)	Melting point, ^{o}C
Aluminum	1.64	.0039	2.7	2.03	28.7	660
Brass	3.9	.002	8.47	1.2	20.2	920
Cadmium	4.4	.0038	8.64	.92	31.6	321
Constantan	28.45	.0002	8.9	.218	14.8	1210
Copper, annealed	1.00	.00393	8.89	-3.88	16.1	1083
Copper, hard drawn	1.03	.00382	8.94	----	----	1083
Copper, beryllium	2.8-3.4	----	8.24	----	----	----
Germ, silver	16.9	.00027	8.7	.32	18.4	1110
Gold	1.416	----	19.32	.296	14.3	1063
Iron (pure)	5.6	.0052-.0062	7.86	.67	12.1	1535
Magnesium	2.67	.004	1.74	1.58	29.8	651
Monel	27.8	.002	8.8	.25	16.3	1300-1350
Nickel	5.05	.0047	8.9	-.6	15.5	1455
Nickel silver	16.0	.00026	6.72	-.33	----	1110

(Table continues to next page)

TABLE 13 -- Concluded

CONNECTOR MATERIALS (ref. 11)

Materials	Relative resistance	Temperature coeff. of resistivity (part per deg. C)	Specific gravity	Coefficient of thermal conductivity (x 10^{-4} per deg. C)	Coefficient of thermal expansion (x 10^{-6} per deg. C)	Melting point, ^{o}C
Phosphor bronze	5.45	.003	8.9	.82	16.8	1050
Platinum	6.16	.003	21.4	.695	9.0	1774
Rhodium	2.6	----	----	----	----	----
Silver	.95	.0038	10.5	-4.19	18.8	960.5
Steel, SAE 1045	7.6-12.7	----	7.8	-.59	15.0	1480
Steel, stainless	52.8	----	7.9	.163	19.1	1410
Tin	6.7	.0042	7.3	-.64	26.9	231.9
Zinc	3.4	.0037	7.14	-1.12	26.3	419

16	Elimination of soldered connections?
17	All of the components necessary for complete assembly delivered in a clear, sealed, dated package?
18	Nonconducting exterior surface?
19	Operating force always in direction of plug travel?
20	Coupling and/or contact seal before electrical contact?
21	Multiple insert "clocking"?

Basic Consideration For Distribution Subsytems

Multiphase electrical power systems for large contemporary aircraft require considerable analytical study to determine the characteristics of the power supply, the distribution circuits, and the protective devices under a variety of operating conditions. A three-phase system can be analyzed by applying ordinary single-phase methods when the system is perfectly balanced. In practice, however, a system is never truly balanced.

Serious unbalance of the system can be caused by many possible disturbances, in which case ordinary analytical methods prove inadequate. The method of symmetrical components, however, has proved very useful and its application is widely employed. This method utilizes the principle that an unbalanced set of vectors, representing currents, voltage or impedances, may be replaced by three simultaneously balanced sets of vectors. In a three-phase, four-wire system these new sets of vectors are known as the positive, negative, and zero sequence components. Each set has its own characteristic voltages, currents, and impedances. Together they enable the solution of an otherwise cumbersome problem.

To facilitate calculations on three-phase, 400-Hz aircraft systems, data for positive, negative and zero sequence impedances on distribution wiring are available. Representative data of a typical installation configuration are presented in table 14.

Effect of Wire Installation on Impedances.-- The impedance will vary with the configuration, spacing, elevation, number of wires per phase and type of neutral return as described below.

(1) Configuration--Affects circuit symmetry which may cause deviation between phase impedances. Utilizing the skin as a neutral return precludes the possibility of a symmetrical installation. A laced group (equilateral configuration) approaches this condition most closely.

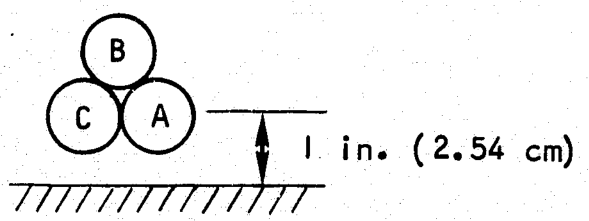
TABLE 14

ONE LACED GROUP, 1-IN. (2.54 CM) ELEVATION

Size	Positive and negative sequence impedance R + jx		Zero sequence impedance R + jx		Single-phase impedance R + jx	
AL-0	.1600	.1941	.3766	1.0377	.2322	.4753
AL-2	.2510	.2004	.4722	1.1360	.3247	.5123
AN-0	.1017	.1840	.3181	1.0367	.1738	.4682
AN-2	.1600	.1938	.3791	1.1069	.2324	.4982
AN-4	.2579	.2010	.4711	1.2090	.3296	.5370
AN-6	.4092	.2196	.6427	1.2633	.4870	.5675
AN-8	.6150	.2210	.8940	1.3570	.7147	.5997
AN-10	1.0144	.2273	1.2370	1.4625	1.0886	.6390
AN-12	1.5200	.2400	1.7100	1.5520	1.5833	.6773
AN-14	2.5016	.2613	2.7360	1.6500	2.5797	.7242
AN-16	4.3600	.2830	4.6100	1.7450	4.4433	.7703
AN-18	5.5373	.2857	5.7677	1.8096	5.6141	.7937
AN-20	6.8000	.3096	7.1800	1.8920	6.9267	.8371

Values in ohms per 1000 ft (ohms per 30.5 meter) at 20°C

1-in. (2.54 cm) elevation



(2) Spacing.--Affects directly circuit reactance to positive (or negative) sequence currents. Tight groupings provide the lowest reactances to positive and negative phase impedances; they reduce losses and interference effects by minimizing eventual coupling of the power circuit with the structure or with other circuits.

(3) Elevation--Affects circuit reactance to zero sequence currents. The zero sequence phase impedance increases with higher elevation. In addition to reducing zero sequence phase impedance, low elevations reduce inductive coupling with adjacent circuits. To a lesser extent low elevations increase losses induced in the structure by positive and negative currents in the circuit.

(4) Number of wires per phase--Multiple circuit configurations offer possibilities of improving the efficiency of utilization of the conductor material. However, this is offset by difficulties encountered in properly protecting such circuits.

(5) Neutral wire return--Installation of a neutral wire return in the same bundle with the line conductors, reduces the zero sequence reactance considerably. Simultaneously it greatly increases zero sequence resistance over that value which would be obtained with skin return. If the skin is basically an aluminum structure a neutral return wire of a size greater than AN-6 in a tight wire bundle will reduce the zero sequence impedance. Smaller wire sizes, however, will increase it.

Characteristic Impedance.-- It is desirable to know the circuit parameters that determine the characteristic or surge impedance of the distribution system and interconnecting wiring. Electrical noise and interference signals generated at one point in the system may be transmitted via the distribution system to many other utilization and communication equipment thereby causing disturbance. Data which are necessary in determining the magnitude and propagation of these signals were assembled in tabulated form. (See table 14.)

Definition: The constant Z_o which is equal to the amplitude or rms value of the incident voltage wave divided by the incident current wave, is called the surge impedance; its reciprocal is called the surge admittance. It is a function of the frequency and the constants r, L, g and C of the circuit.

$$Z_o = \frac{Z}{Y}$$

where $Z = r + j\omega L$, complex series impedance of the conductor

$Y = g + j\omega C$, complex shunt admittance of the conductor

and r = resistance

L = inductance

g = conductance of the shunt leakage path

C = capactiance to ground

These values are in ohms, henries, mhos and farads per unit length of the conductor. The surge impedance is therefore independent of the wire length.

Parameters determining surge impedance: For all practical considerations, it can be assumed that the leakage conductance is equal to zero and the frequency of the transient is high enough (50 kHz or higher) so that the formula for the surge impedance simplifies to:

$$Z_o = \sqrt{\frac{L}{C}}$$

Table 15 shows the aircraft wiring parameters L and C installed at various distances from the ground plane (airplane skin). If a wire is routed in a bundle, the factors L and C vary considerably depending on the number of adjacent wires in the bundle and the presence of current and its direction in the neighboring wires.

To find accurate values of surge impedance of particular distribution branches by theoretical analysis is a complex and complicated operation. It may invalidate the results if too many assumptions have to be made. For rough approximations of the surge impedance of wires routed in bundles, the single wire values from the table may be used. In this case the single wire data may deviate by ±20 percent.

Factors affecting surge impedance: Surge impedance is affected by the factors listed below.

(1) Frequency or rate of rise of the transient voltage (in all practical cases this can be assumed to be higher than 50,000 Hz)

(2) Size of conductors (heavier sizes have lower surge impedance)

(3) Spacing of the conductor to the airplane skin structure (closer spacing decreases the surge impedance)

(4) Presence of other conductors in close proximity (this decreases surge impedance)

(5) Number of neighboring conductors (more wire in the bundles decreases surge impedance)

(6) Size of neighboring conductors (heavier size wires in the bundle decrease surge impedance)

TABLE 15

SURGE IMPEDANCE PARAMETERS
(English unit only)

Wire size (AWG)	Spacing, in.	L $\mu\mu$ H/in.	C $\mu\mu$ F/in.	Z_o Surge impedance, ohm
22	1.0	24370	.294	287
	1.5	26430	.271	312
	2.0	27890	.257	329
	2.5	29000	.246	342
	3.0	29950	.239	353
20	1.0	23260	.308	274
	1.5	25230	.283	299
	2.0	26970	.267	316
	2.5	27920	.256	329
	3.0	28850	.248	340
18	1.0	22000	.324	260
	1.5	24120	.297	284
	2.0	25580	.280	302
	2.5	26716	.268	315
	3.0	27640	.259	326
16	1.0	21250	.337	250
	1.5	23310	.307	275
	2.0	24770	.289	292
	2.5	25900	.276	305
	3.0	26830	.267	316
14	1.0	20130	.356	237
	1.5	22190	.322	262
	2.0	23650	.303	279
	2.5	24790	.289	292
	3.0	25710	.278	303
12	1.0	18950	.378	223
	1.5	21000	.341	248
	2.0	22470	.319	265
	2.5	23600	.303	278
	3.0	24530	.292	289
10	1.0	17480	.409	206
	1.5	19540	.366	230
	2.0	21000	.341	248
	2.5	22140	.323	261
	3.0	23060	.310	272

TABLE 15. -- Concluded

SURGE IMPEDANCE PARAMETERS
(English unit only)

Wire size (AWG)	Spacing, in.	L µµ H/in.	C µµ F/in.	Z_o Surge impedance, ohms
8	1.0	15870	.451	187
	1.5	17930	.399	211
	2.0	19390	.369	229
	2.5	20520	.349	242
	3.0	21450	.334	252
6	1.0	14780	.484	174
	1.5	16840	.425	198
	2.0	18300	.391	216
	2.5	19430	.368	229
	3.0	20360	.352	240
4	1.0	13660	.524	161
	1.5	15720	.456	185
	2.0	17180	.471	202
	2.5	18310	.391	216
	3.0	19240	.372	227
2	1.0	12450	.575	147
	1.5	14570	.494	171
	2.0	15970	.448	188
	2.5	17100	.419	202
	3.0	18030	.397	212

(7) Presence of current in adjacent conductors (this will generally lower the surge impedance)

(8) Direction of current in neighboring conductors (this can alter the surge impedance both ways, same direction of current flow in adjacent wire decreases, opposite direction increases the surge impedance)

Corona.--Because of the increasing altitude and environmental requirements of modern aircraft, the influence of corona on electrical power generation and distribution systems necessitates additional consideration in system design. Electrical equipment installed in unconditioned and unpressurized areas will be subjected to the likelihood of corona discharge between parts and different electrical potentials. Because corona is most undesirable on aircraft, suitable design action has to be taken for its prevention. A description of corona is given below.

Definition: Corona is an audible, luminous electrical discharge due to ionization of the insulating gas surrounding an electrical conductor. Electrical equipment should be designed or protected so that corona discharge does not occur.

Undesirable Effects: Among the undesirable effects of corona are:

(1) Radio frequency interference--Corona discharge current has component frequencies over a wide band to 800 MHz and causes a serious radio interference problem. Besides, this interference is fed to all utilization equipment through the power feeders.

(2) Insulation breakdown--Corona discharge is the result of gaseous insulation being overstressed. A further increase in electrical stress will result in insulation breakdown and arc-over which is a fire hazard. Ozone, a chemically active gas, is usually released in this discharge; many insulating materials and metals deteriorate very rapidly in this atmosphere.

(3) Power loss--A sustained corona discharge consumes electrical energy and dissipates it as heat to the environments. This power loss is undesirable.

Factors affecting corona: Some of the many factors affecting corona are closely interwoven, forming complex dependencies which are not completely understood. The following is a qualitative discussion of some of the factors.

(1) Pressure--Corona breakdown voltage decreases with decreasing pressure to a certain level in the extremely low pressure region and then increases as the pressure further decreases. Thus, the corona onset voltage is decreased to a substantially low value for altitudes above 50,000 (15,200 miles). Paschen's Law states that the breakdown voltage varies as the product of the pressure and the electrode spacing. This is illustrated in figs. 45 and 46. However, departures from this law can be large for nonuniform fluids and for very large or small gaps.

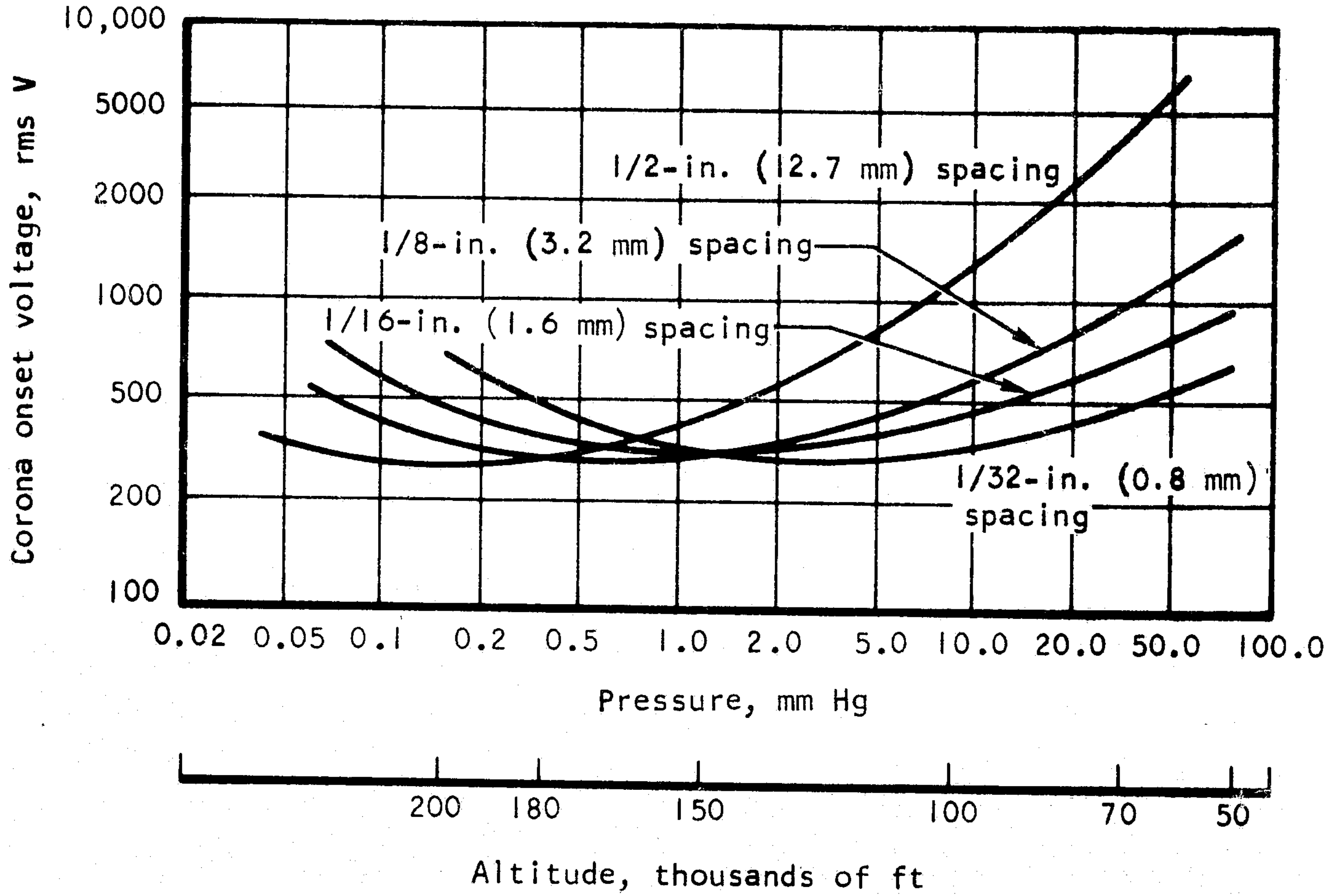


Figure 45. Corona Onset Voltage as a Function of Pressure or Altitude for Various Spacings

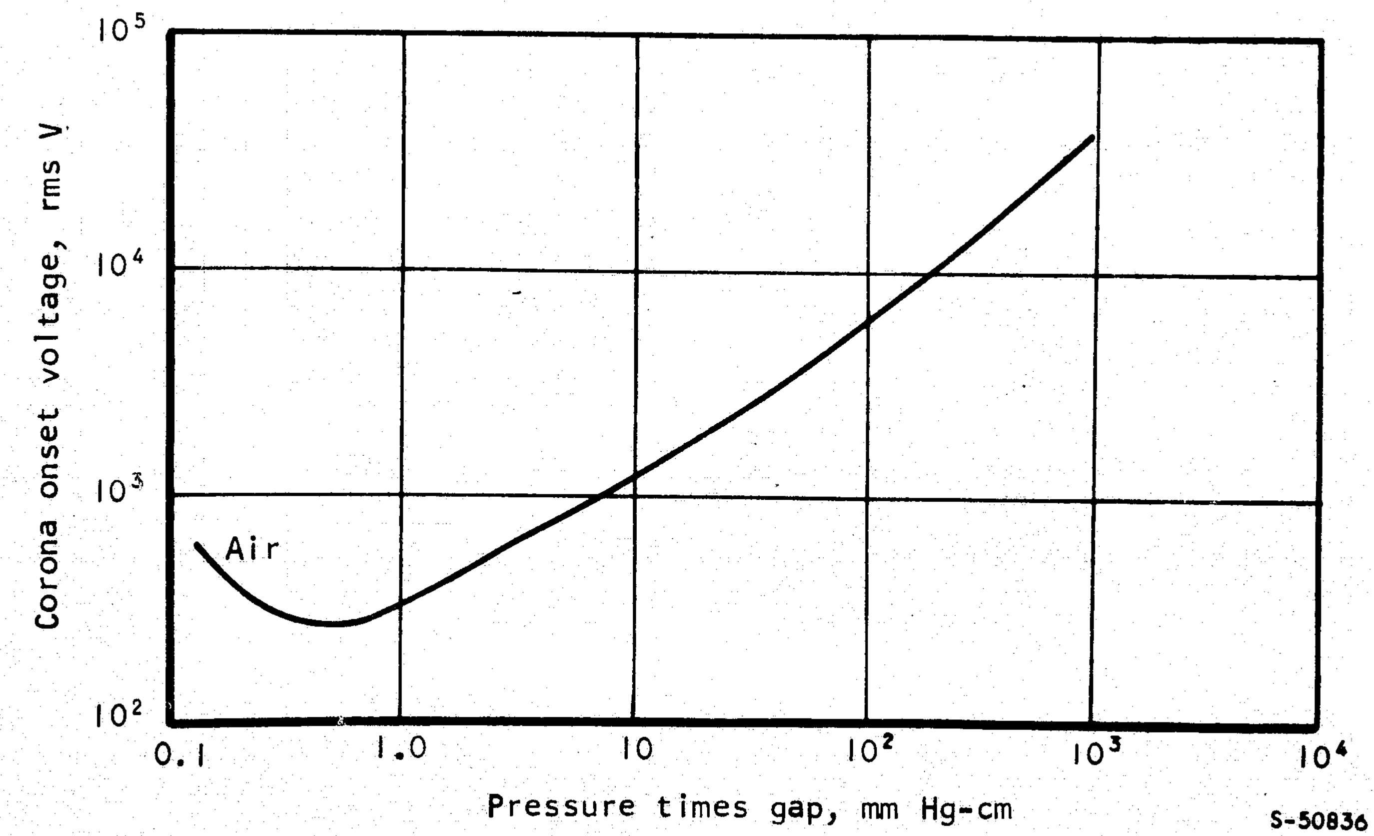


Figure 46. Corona Onset Voltage in Air as a Function of Pressure-Spacing

(2) Insulation materials--Corona breakdown voltage for solid insulations is much higher than that of gases since gases are easy to ionize. Thus, corona in solid dielectrics occurs only in the air surrounding the solid or in voids within the solid. Various insulation materials have different breakdown voltages. Fig. 47 illustrates the change in minimum onset voltage for several wire insulation materials and wire sizes.

(3) Temperature--The effect of temperature is similar to that of pressure because the gas pressure varies proportionally with temperature. The actual effect can be calculated by using the gas law. However, at high temperature, electrodes of certain materials begin to vaporize and the corona onset voltage decreases drastically.

(4) Frequency--The corona onset voltage of a gap is approximately the same for common ac power frequencies (50 and 60 Hz) as it is for the dc voltage. As the frequency is increased, the corona voltage decreases. This effect becomes noticeable with 400-Hz power for gaps greater than 1 cm. It is more difficult to insulate and eliminate corona for ac voltages than for dc voltages. However, once leakage current is started, corona breakdown action on the insulation is faster in dc than in ac voltages. Under some conditions, dc corona may exist, apparently without generation of rf noise.

Secondary factors: Other factors affecting the corona onset voltage are the configuration of the electrode, voltage gradient, radiation, etc.

Corona-safe voltage levels: Generally speaking, a voltage of 300 V is considered safe from corona in unpressurized areas of the aircraft in high altitudes up to 100,000 ft (30,480 meters). In pressurized areas, the minimum corona onset voltage for a bare conductor is about 700 V.

Insulation: Dc voltage is more difficult to insulate than ac voltage. In dc systems, once a leakage current exists, corrosive action builds up faster since it is not hampered by the continuous change in the polarity. Dc voltages are more likely to cause wire fire than the equivalent ac potentials; dc voltages do not, however, produce the heavy arcing at the fault that ac voltages do.

Solid-state components: The forward drop of solid-state power components is approximately constant regardless of the current. The weight of the component increases with the current rating. Therefore, for better efficiency and lower weight, the system voltage should be as high as practical. The peak inverse voltage ratings (PIV) of present day diodes, thyristors, and transistors are higher than the voltage levels considered as safe for aircraft operation.

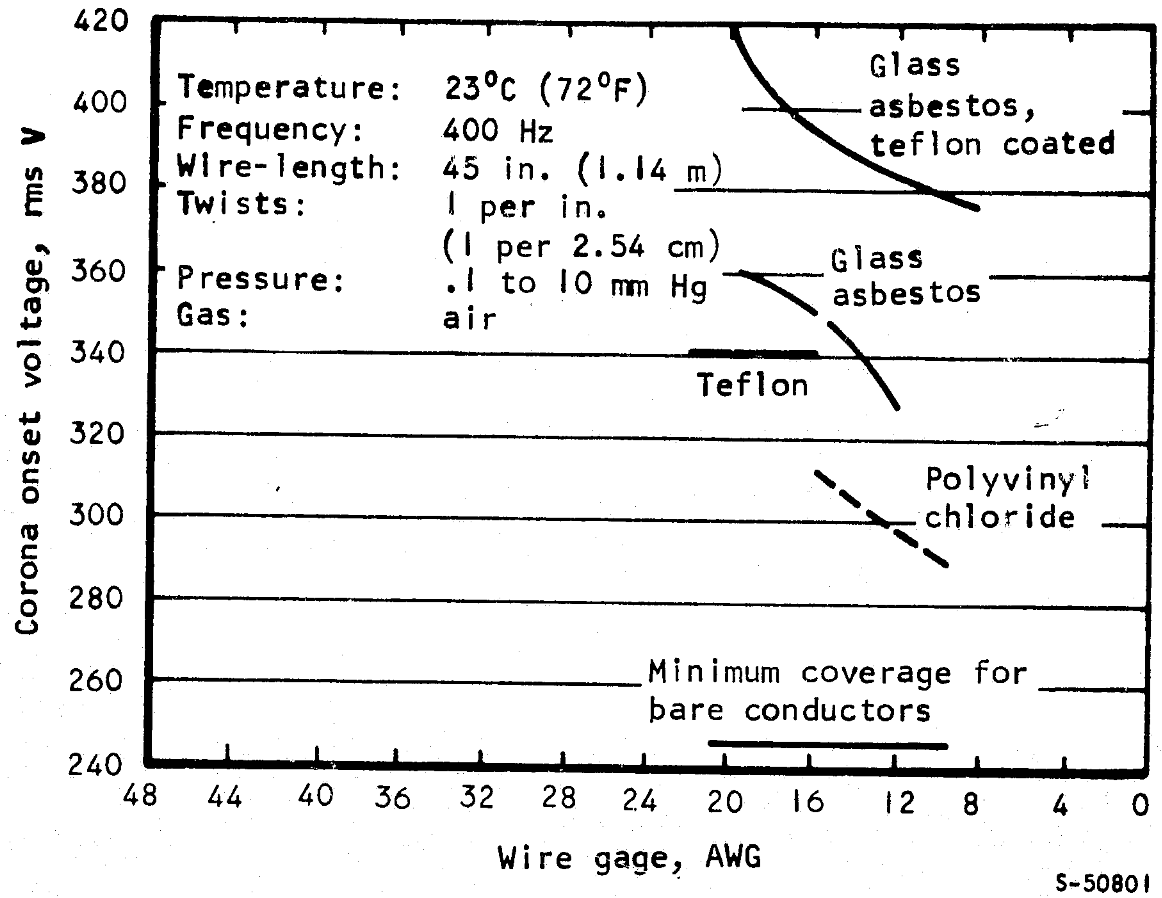


Figure 47. Minimum Corona Onset Voltage (COV) of Several Wire Insulation Types and Wire Sizes

GENERATION SUBSYSTEM

Introduction

Numerous aircraft generating systems are in use today, many of them are quite old and still utilize one or more 28-Vdc generators, which are connected to a common busbar. These old generators were shunt-wound, compensated commutator-brush generators. Later, 3-phase 115/200 V wild frequency generators in combination with dc systems found expanded usage. The generating system used most frequently in modern aircraft and discussed in this section is the constant speed, constant frequency (CSCF), 400-Hz 115/200-V 3-phase ac system.

A logical method of obtaining energy for the aircraft electrical system is to extract mechanical shaft power from the propulsion engines. Since the electrical demand is insignificantly small (in the order of 1 percent of engine capacity) in comparison to the propulsion power, the prime mover represents a stiff source to the electrical generating subsystem.

Aside from reliability, the most important factor in the generating subsystem is weight per kilowatt of generated power. Designers are constantly striving either to reduce weight or increase output while keeping weight stable. They are also trying to package their end product in the smallest possible space. The weight of the generating equipment with accessories for a large contemporary four engine subsonic transport airplane is approximately 700 lb (320 kg). This is 175 lb (80 kg) of electrical installation per generating channel and includes the constant speed drive (CSD), generator, heat exchanger, and the necessary plumbing and air ducting. The speed of the generator is typically 6000 rpm and the method of cooling is airblast. Large transport airplanes presently under construction, however, will install electrical generating plants which are nearly 50 percent lighter. This substantial weight reduction was made possible by a new design concept, the integral drive generator (IDG) which combines the CSD and generator into one housing. The higher operating speed (12,000 rpm) and the unique method of oil splash cooling contributed significantly to the weight reduction.

The CSCF system remains the standard approach for obtaining constant frequency by turning a synchronous alternator at constant speed. In the last 15 years, there have been significant improvements in the development of constant speed drives with respect to weight and performance. A block diagram showing the conventional aircraft generation subsystem is shown in fig. 48.

As shown in fig. 48, the output speed of the CSD is controlled by both the frequency signal and the real load sharing signal; the generator field excitation is controlled by both the voltage signal and the reactive load sharing signal. The PMG supplies all the control power and the frequency signal. The voltage response of this brushless generator is slow because the exciter field and main generator field cause two time delays. The exciter is necessary in this case to accomplish a brushless construction.

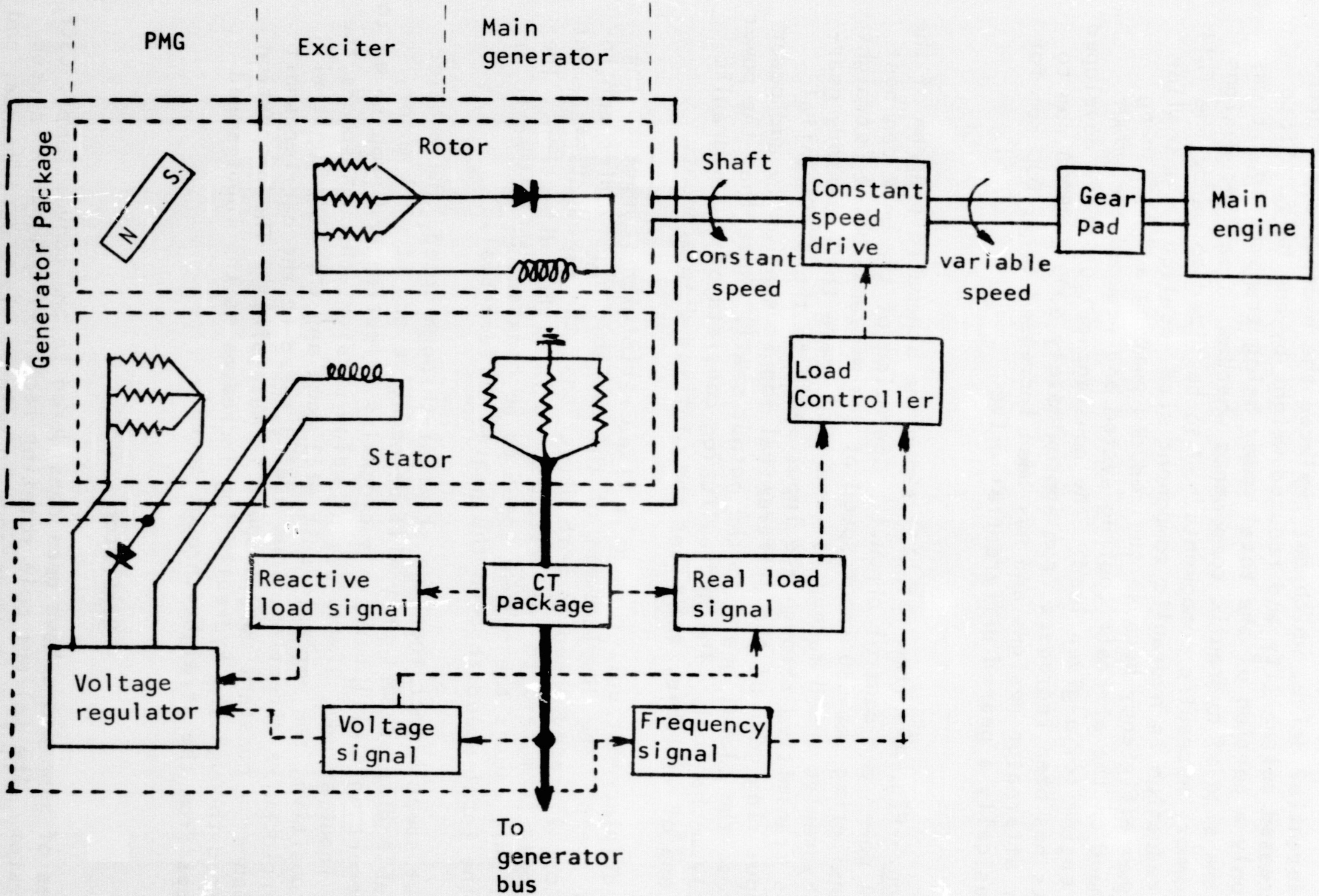


Figure 48. Aircraft Generation Subsystem

Constant Speed Drive

The geared differential drive, which has replaced the hydraulic differential drive, has increased reliability and reduced weight by one half. Since hydraulic power is only a portion of the total power being transmitted, it was possible to reduce the size of hydraulic components whereas in earlier designs, using a hydraulic summer, hydraulic components had to be sized to carry the full rated torque. The reduction in hydraulic component size resulted in a smaller power train and higher efficiency because the reduced pumping and thrashing losses put less demand on the aircraft cooling system and reduced the power extracted from the engine for a given load. The constant output speed developed by these drive units has been increased from approximately 6000 to 8000 rpm to accommodate lighter alternator designs and has been increased to 12,000 rpm for the IDG which is basically a geared differential drive.

Basic Operation.--Fig. 49 (from ref. 12) shows the schematic diagram of the geared differential power and control circuit. Operation of the unit can best be described by considering the CSD input speed at the midrange point (straight through) where no hydraulic speed makeup is required. Here the accessory gearbox shaft output is transmitted through the input shaft to the differential gear unit. The input ring gear of the differential log is stationary and power is transmitted across the planet gears to the output shaft. Note that the power transfer is strictly mechanical. The hydraulic log consists of two hydraulic pump/motor units, one of variable, the other one of fixed displacement.

The variable displacement unit runs at a fixed ratio with respect to the transmission input speed. The displacement of the variable unit varies continuously from zero to full rated speed in both directions; the fixed displacement unit runs at any speed from zero to full rated speed in either direction. The fixed displacement unit causes the output speed to be constant by adding to or subtracting from the input speed through the geared differential.

With the input speed higher than the straight-through speed, the wobbler angle of the variable displacement unit will be negative and it will motor. The fixed displacement unit will turn in a clockwise direction and pump so as to subtract from the input speed. Similarly, at below straight-through speeds the wobbler angle is positive, the variable unit will pump and the fixed unit will motor in a counterclockwise direction, thus adding speed to the carrier shaft speed. In this manner a constant output speed is maintained. The proportion of power transmitted through the hydraulic log increases as the input speed deviation increases from the straight-through speed.

Generator

Various types of generators have been considered in the conventional, CSCF electric power system. The salient pole rotating rectifier generator is used most often because of its inherent advantages in weight, size, performance, and efficiency. Other types of generators offer the advantage of eliminating rotating rectifiers as well as rotating windings. Some typical solid rotor generators are the Bekey-Robinson-Lundell, the doubled-ended inductor, the Rice, and the

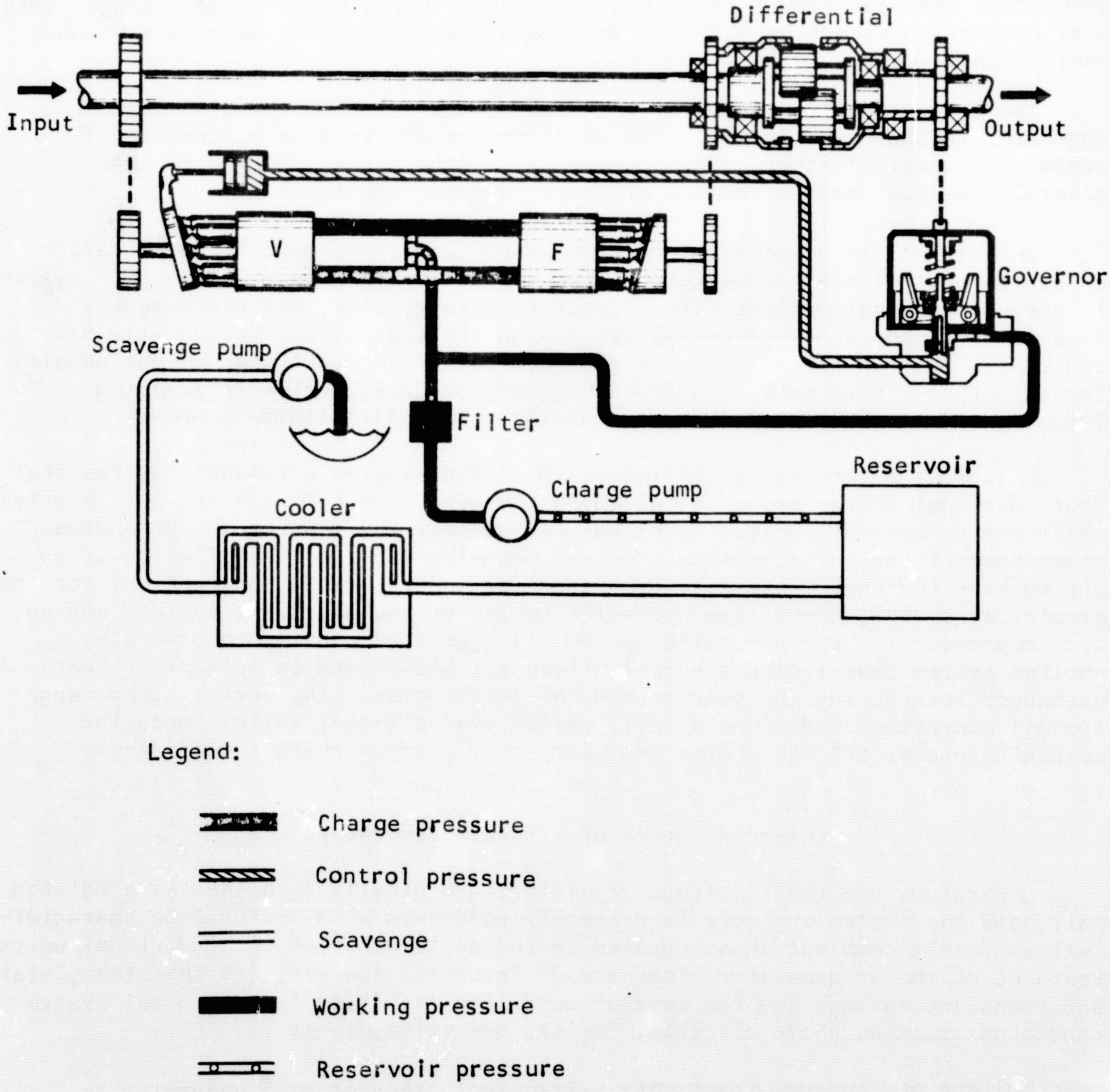


Figure 49. Geared Differential Transmission Schematic (ref. 12)

Inductor-Lundell-inductor. These types, however, are longer, heavier, and less efficient; they exhibit poorer performance because their leakage flux and leakage reactance are higher than that of the conventional machine. Solid rotor generators are also slower in voltage recovery after a sudden load change. They are, however, best suited for high-speed applications or for extreme environmental conditions.

The representative weight of air cooled, 400 Hz, 8000 rpm aircraft generator is shown in fig. 50. The generator weight is also a function of speed and number of poles. The approximate weight variation of a 60 KVA generator versus speed or number of poles is shown in fig. 51.

Unless extreme temperature or high speed conditions make the application of solid rotor generators mandatory, the best suited generator for a CSCF system is the conventional machine with respect to weight, size, and performance. Because this generator approaches its design limit at 22,000 ft/min (112 m/sec) top speed, its rating and size are confined unless an additional weight penalty for structure enforcement or odd shape generator construction is accepted. Fig. 52 indicates a broad limit for generator operating speed vs rating.

A typical modern day installation for a four-engine airplane requires that four identical gearboxes be installed in the wing, one for each engine. A drive shaft and a power disconnect coupling will connect the gearbox to the engine power takeoff pad. The gearbox provides mounting pads for installation of an air turbine for engine start, a generator (or generator-CSD combination) for the electrical system, one or two hydraulic pumps for the hydraulic system, and an air compressor for the pneumatic system. The gearbox usually incorporates a cooling system that includes a gear-driven air blower and an oil-to-air heat exchanger for cooling the gearbox and the accessories. The engine speed range for all operating conditions usually varies over a 2-to-1 ratio, hence the mechanical power off the engine accessory gear gearbox changes accordingly.

Characteristics of Aircraft Generators

Generators and their voltage regulators are usually furnished as a matched pair, and the system engineer is primarily concerned with performance characteristics of this combination as opposed to the performance of the individual units. Features of the ac generator, that are of interest, however, are the steady state and transient ratings and the type of operation resulting from abnormal system conditions such as short circuits, faults, overvoltages, etc.

Voltage and current transients.--Transient behavior of a generator is shown in fig. 53. Transient voltage due to a sudden load application and load removal are shown in figs. 53a and 53b, respectively. In the curves, AB is approximately equal to the product of transient reactance of the generator and the reactive current of the suddenly changing load. Fig. 53c shows the short circuit current transient with constant and regulated excitation.

Machine ratings.--The capability of a machine or group of machines to carry load in a system depends upon two types of limits, stability and thermal. The stability limits can be divided into two categories: steady-state and transient. The transient stability can be further subdivided into two cases:

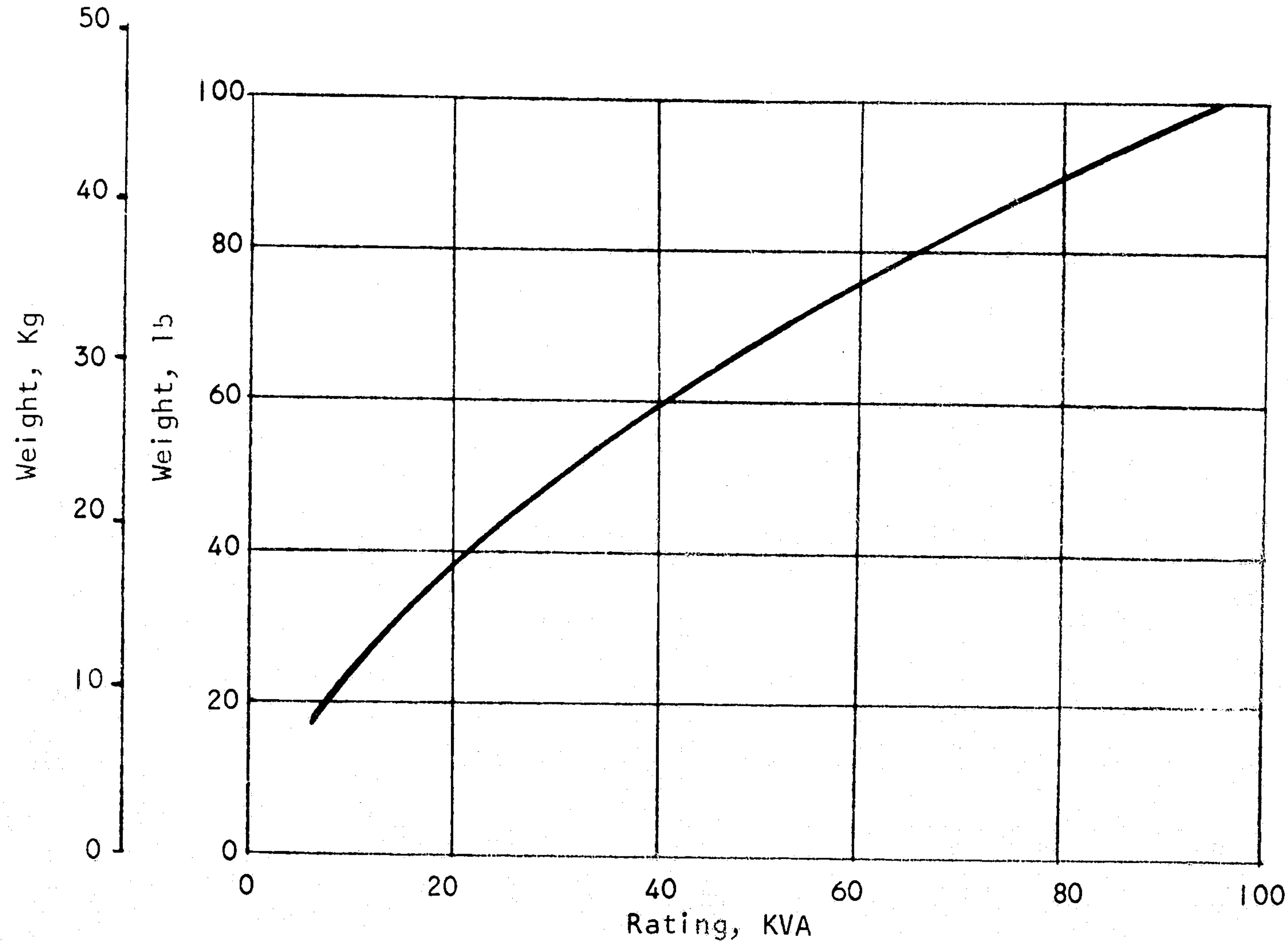


Figure 50. 400 Hz, 8000 rpm, Air Cooled Aircraft Generator Weight

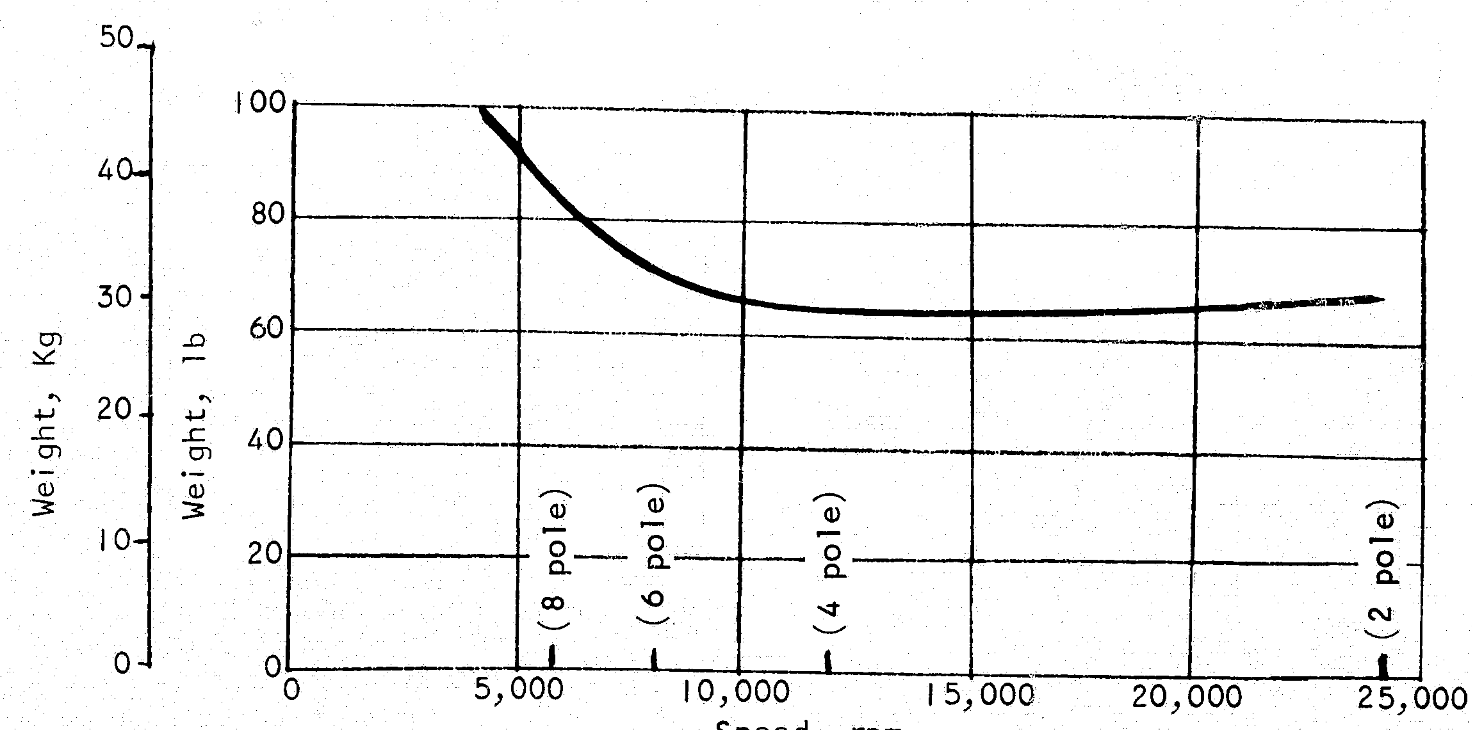


Figure 51. 400 Hz, 60 KVA, Air Cooled Generator
Weight vs Speed

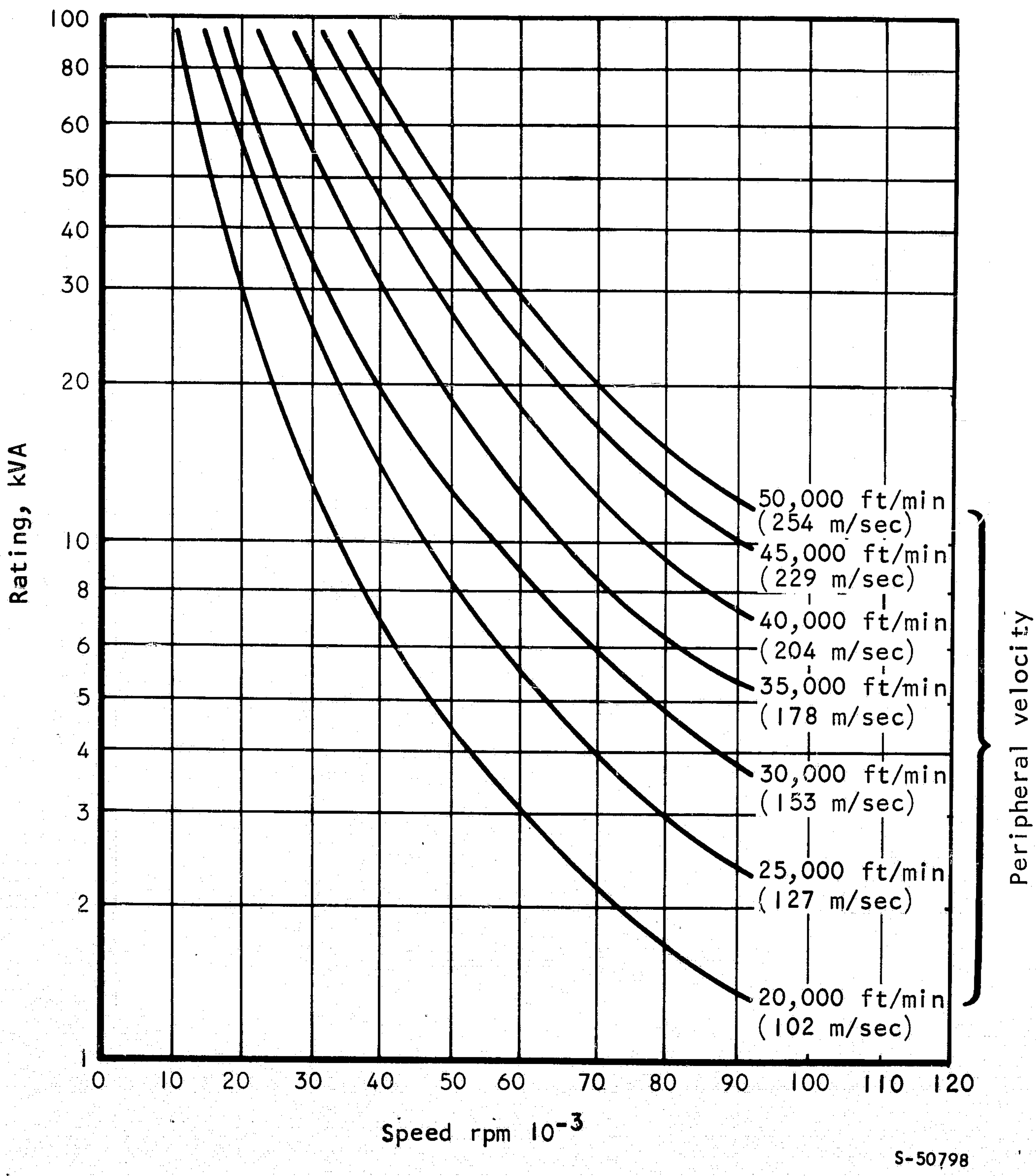
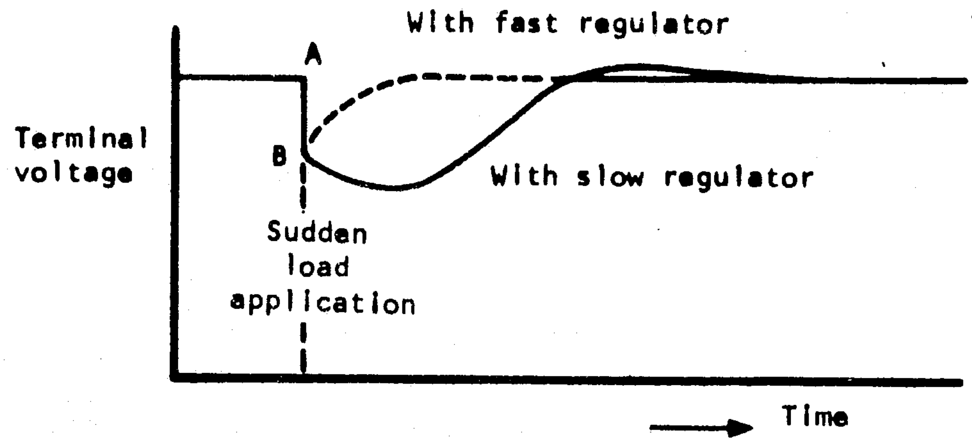
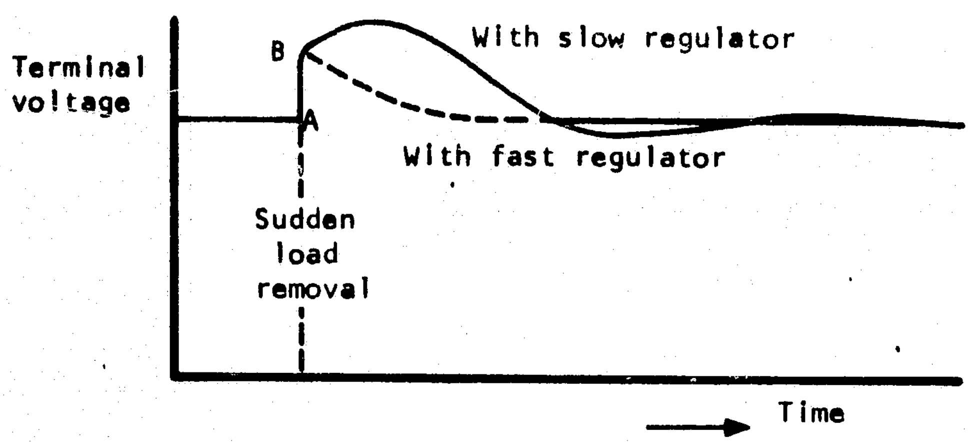


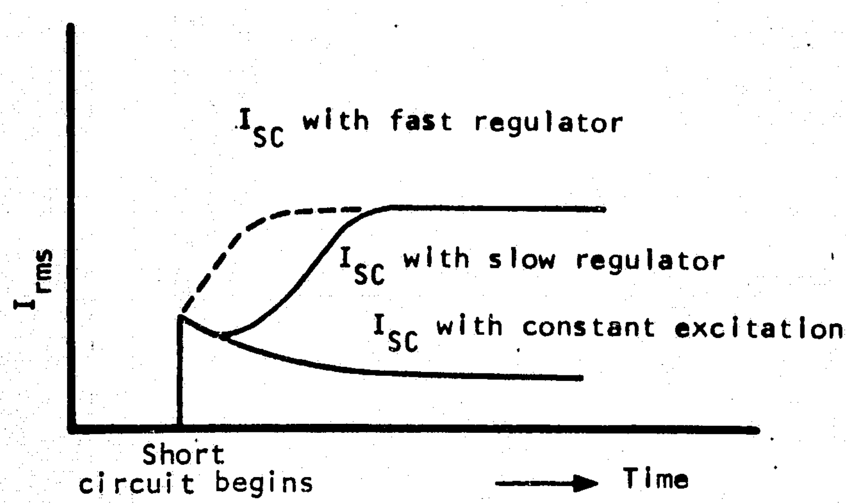
Figure 52. Alternator Rotating Speed, Tip Speed and Power Output Relationship



a. Voltage transient, sudden load application



b. Voltage transient, sudden load removal



c. Alternating-current generator fault currents with constant and with regulated excitation

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Figure 53. Generator Transient Behavior

(1) finite machine or machines connected to an infinite bus; and (2) two or more finite machines connected together. The latter case is also referred to as hunting stability. Steady-state stability concerns the power limit under slow or gradual load changes. The transient stability is the power limit under transient or sudden load changes.

A "rule of thumb" for determining steady-state power limit is that it is equal, roughly, to the three-phase short-circuit current in per unit at ceiling excitation.

The transient stability limit is small than the steady-state power limit. In an aircraft electrical power system, two or more small generators are connected in parallel. Since these generators are not far apart in distance, and there is appreciable resistance in the generator armature and generator feed, hunting stability is not a problem in the aircraft power system.

Military specification MIL-G-6099A has established minimum thermal limit standards for aircraft alternating-current generators. It provides three classes of thermal ratings for generators and regulators. Class A rating has an inlet air temperature of 40°C. Class B rating has an inlet air temperature of 80°C; and class C has an inlet air temperature of 120°C. Limitations are specified on the mass flow of air, and the pressure drop across the machine at all ratings.

For the conventional wound rotor generator, in low altitudes (below 20,000 ft) the stator winding temperature usually determines the thermal limit. For high altitude operation (above 20,000 ft) the rotor is usually reaching the thermal limit first. Therefore, to improve altitude performance, better cooling should be provided for the rotor. On the other hand, if sea-level performance improvement is desired, the generator should be designed with more stator copper and better cooling paths around the stator.

Fault power requirements.--The power consumed by an ac machine and the system under a fault depends on the type of fault and the machine constants.

If a 3-phase fault with zero resistance is placed on the generator terminals, the actual power required is equal to the sum of the armature I^2R losses (where I = armature current and R = armature circuit resistance), the field $I_f^2 R_f$ (where I_f = field current and R_f = field circuit resistance) and the friction and windage losses. When fault resistance and line losses are added, the power required becomes appreciable.

If an unbalanced fault is placed on the machine, two additional power requirements are present: (1) Additional field windings, such as ammortisseur bars, result in I^2R losses that increase the machine losses, and (2) the power required by loads on the machine, before the fault occurred is effected. With a static-type load, this power requirement will increase even further since the voltage is increased on the unfaulted phases. On typical systems which have been analyzed, single-phase or line-to-line faults can result in requirements of 2 to 2.5 per unit power.

Overvoltages.--Transient overvoltage is usually resulted from a sudden rejection of a heavy load or clearing of a short circuit. A transient overvoltage of 1.7 per unit following a sudden removal of a short circuit was observed for a 40-kVA constant-speed machine.

Steady overvoltage can be caused by one of the following conditions:

(1) A single-phase fault causing a steady-state overvoltage on the unfaulted phases

(2) Malfunction of the excitation system causing the exciter to go to ceiling

(3) An open phase on an isolated generator

The steady-state voltages on the unfaulted phases during a single-phase fault will be somewhat lower than transient overvoltages and for the above mentioned 40-kVA constant-speed machine are approximately 1.5 per unit. Since the machines are built to have low unbalanced voltages for unbalanced loads, this condition has been minimized.

Overvoltages on isolated generators due to excitation system malfunctions are limited only by saturation and available field current. In a parallel system, however, bus overvoltage will generally be lower due to the action of the normal generators on the system.

Overvoltages on isolated generators due to an open phase result from the method of sensing in the regulator which will attempt to hold the average voltage of the two remaining phases equal to the normal 3-phase average.

Emergency Electrical Power Supplies in Existing Aircraft

The source of emergency electrical power in existing aircraft is either a battery or an auxiliary electric generator driven by various type prime movers. A brief review of these sources is given in this section.

Aircraft batteries.--Aircraft batteries are used to provide standby sources of power and, in small airplanes, also to start the engine. In case of generator failure, the batteries will supply power to radio communication equipment, instruments, essential lighting, and controls for sufficient time to allow the aircraft to land. The batteries are kept fully charged by the main generators during flight. Batteries selected for aircraft application should be lightweight, low cost, rugged, nonspill, tolerant of wide changes in temperature and pressure, and capable of long charge retention and rapid recharge from the normal aircraft charging system.

If engine starting is a requirement for the battery, the battery should be designed with very thin plates, so that very high, instantaneous performance can be obtained at low weight.

Three types of batteries are used in aircraft today.

(1) Lead acid battery

(2) Nickel cadmium battery

(3) Silver-oxide zinc battery

Performance data of these batteries are given in table 16.

Lead acid batteries: The lead-acid system has the advantages of high voltages, good cycle life, and low cost. Its disadvantages are weight, bulk (low energy density), poor capacity retention, and low temperature service. The aircraft lead acid batteries are similar to those batteries for automotive starting. Its electrodes are cast lead-antimony grids filled with a paste of lead oxide, water, and sulfuric acid. The paste of the negative plates contains about one percent of expander, which is composed of barium sulfate and an organic material, such as lignin.

The barium sulfate provides a surface for the deposition of lead sulfate, and the lignin prevents it from crystallizing on the sponge lead surface, thereby improving the high discharge and low temperature characteristics of the batteries. After the plates have been formed, dried, and assembled, they are immersed in a weak solution of sulfuric acid and charged. The lead oxide of the negative plate is reduced to sponge lead, and the lead dioxide is formed at the positive plates.

Recent progress in lead acid batteries is achieved by using thinner plates, thereby increasing surface area for the same weight. The increase in surface area enables a reduction in working surface current density, resulting in lower internal resistance. This, in turn, reduces the internal I^2R loss and the internal voltage drop.

Other recent improvements of lead acid batteries include the use of a lightweight polystyrene container and the arrangement of passing the intercell connectors through the cell walls. Both of these help to reduce the weight of batteries.

Nickel-cadmium batteries: The nickel-cadmium (Ni-Cd) battery is mechanically rugged and capable of withstanding electrochemical abuse such as overdischarging, overcharging, and standing idle in the discharged state. Although Ni-Cd batteries operate at slightly lower voltages, they exhibit very little self-discharge and can be used over a temperature range of -55° to 80°C. These batteries evolve practically no gas on discharge or idle stand so that for some purposes they can be hermetically sealed. In addition, Ni-Cd batteries are characterized by good cycle life and can be kept fully charged by a very small current (floating charge), resulting in little water consumption. By changing the construction of the plates, it is possible to build batteries for low or high rate applications. The positive and negative plates are of the same mechanical design and external appearance.

TABLE 16

PERFORMANCE DATA OF BATTERIES IN EXISTING AIRCRAFT

Characteristic	Cell type		
	Lead acid (Open rectangular)	Nickle-cadmium (4 AH sealed cylindrical)	Silver-oxide zinc (open rectangular)
Energy density			
Theoretical	55 W-hr/lb (121 W-hr/kg)	95 W-hr/lb (209 W-hr/kg)	196 W-hr/lb (432 W-hr/kg)
Reported per unit weight	10 to 20 W-hr/lb (22 to 44 W-hr/kg)	12 to 14 W-hr/lb (26 to 31 W-hr/kg)	40 to 74 W-hr/lb (88 to 165 W-hr/kg)
Reported per unit volume	1.0 to 2.0 W-hr/cu in. (16 to 33 W-hr/ cm^3)	1.15 to 1.30 W-hr/cu in. (18.8 to 21.3 W-hr/cm^3)	2.0 to 4.8 W-hr/cu in. (33 to 79 W-hr/ cm^3
Cell voltage			
Theoretical	2.05	1.299	1.672*** 1.560****
Open circuit voltage*	2.0 to 2.05	1.30 to 1.35	1.86
C/8 to C/5 rates**	1.95 to 2.05	1.25	1.58
C/2 rate		1.20	1.54
C/0.2 rate		0.95	1.37
Cycle life			
50 percent depth of discharge		2000 to 3000	100 to 300

Notes: *The differences in the open circuit voltage and theoretical voltages may be due to the time the cell stands between charge and discharge.

**C represents the capacity of the cell. C/N rate of discharge indicates a rate at which the nominal capacity of the battery is removed during a N-hour period.

***Value associated with the higher (+2) oxidation state of silver.

****Value associated with the lower (+1) oxidation state of silver.

In one type of construction (pocket type), the active materials are tightly packed in perforated flat tubes or pockets and mounted horizontally in nickel-plated steel frames. The plate separators are rods of insulating material that rest in vertical grooves pressed into the faces of the plates. The active material in the positive plate is initially nickelic hydroxide with about 25-wt percent of a specially treated graphite, the latter acting as a current conductor. The negative material, cadmium, in some instances contains about 25-wt percent iron oxide, which acts as a disperser to prevent the packing together of metallic cadmium particles formed on reduction of cadmium oxide (CdO) during charge. The graphite and iron take no part in the cell reaction, and the electrochemical reactions that occur in the cell during charge and discharge are very complex and not fully understood.

The second type of construction (sintered-plate) uses porous plates made from nickel powder which is molded into shape and sintered (at 900°C) in a reducing atmosphere onto nickel or nickel-plated supports such as screens, perforated metals, or expanded metals. This produces porous plates with pore volumes or porosities in the region of 70 to 90 percent. These plates are then impregnated with active material by soaking them in a solution of nickel salts to form the positive plates or a solution of cadmium salts to form the negative plates.

The pocket-plate design is more rugged, but the sintered-plate design, because of the thinner plate construction and larger surface area, has better high-rate discharge and low-temperature performance characteristics. At low drain rates, the pocket- and sintered-plate batteries have similar characteristics. The Ni-Cd battery is particularly suitable for services where the charge and discharge rates fluctuate considerably. Aircraft Ni-Cd batteries are mainly sintered-plate design. The limitations of Ni-Cd batteries are low energy output per unit weight and volume, moderately high cost, the need for an overcharge to obtain full capacity output, and the time required (12 to 16 hr) to recharge the sealed cells. Rapid charging was possible only through the use of complex chargers or supplementary devices such as auxiliary electrodes that added to weight, volume, and cost. However, recent developments, such as the fast-charge cell and new charging innovations, have made it possible to charge Ni-Cd batteries rapidly and safely without the need for external control devices, reducing the charge time for some applications to approximately 1/2-hour.

Silver-oxide/zinc batteries: The silver-oxide/zinc (AgO-Zn) battery is representative of a new class of secondary batteries that was not commercially available until after World War II. Although the potentialities of this electrochemical system were known for many years, it was not until the early thirties that the AgO-Zn battery became practical. Up until that time, the major difficulty with this system was the intermixing of the active materials during charge and discharge. This problem was later resolved to a considerable extent by interposing a semi-permeable cellulosic membrane separator between the plates.

In these cells in the charged state, the cathode consists of silver oxide electrolytically formed on silver or a silver-plated copper screen. The anode consists of sponge zinc on a suitable collector, usually silver or silver-plated copper foil. The electrolyte is an aqueous solution of potassium hydroxide (KOH) in concentrations over the range of 30 to 45 wt percent. The separator materials used most frequently are cellophanes or various modified cellophanes. Most of the electrolyte in this type of cell is absorbed by the porous electrodes and separator materials. In the fully charged state, the electrolyte level reaches to approximately one-third the height of the plates and reaches all the active surface of the electrodes by capillary action.

On discharge, the silver oxide is eventually reduced to metallic silver and the zinc is oxidized to form the hydroxide. The charge and discharge characteristics are complicated at times by the presence of two distinct voltage plateaus, corresponding to the monovalent and divalent states of silver. This phenomenon is not noticeable at high rates due to polarization. An initial high voltage level is observed at low drain rates and can be eliminated by preloading the cell until the voltage falls to the lower value.

For the combination of high energy density and high power density, the AgO-Zn battery is undoubtedly the leader among present-day practical rechargeable systems. These batteries deliver high currents and give three to five times higher W-hr capacities per unit weight and volume than lead-acid or Ni-Cd batteries. Other advantages of the AgO-Zn battery are good charge retention, high charging efficiency, good voltage regulation, and good high-temperature operating range.

Although the AgO-Zn system may exhibit all the desirable features listed above, it also has the property of being one of the most thermodynamically unstable. Finely divided zinc is highly susceptible to oxidation in the presence of moisture. Fortunately, in the absence of moisture, the rates of the degradation (and passivation) reactions involved are very slow. For this reason, AgO-Zn batteries are usually shipped in dry form and sealed to keep out moisture. Another undesirable characteristic of this system is the high solubility of the electrode reaction products in the electrolyte. This zinc discharge product in particular is highly soluble and does not redeposit in its original form or position on the electrode during charge, resulting in rapid deterioration of performance. Another serious weakness of the zinc electrode is its tendency to form dendrites or "trees," which, if they remain attached to the anode, cause shorting of the cell. Other limitations are short storage and cycle lives, sensitivity to overcharge, poor low-temperature performance, and high initial cost.

The AgO-Zn batteries have not found wide aircraft application because of their short cycle life and the requirement of special charging facilities because they are not suitable for constant-potential charging. High discharge currents can generate high internal temperatures which may damage the separators and possibly even destroy the cells.

Auxiliary Power Units.--Auxiliary power units generate emergency power independently of the aircraft main engines. These units can supply power to the airplane while on the ground, when all engines are out during flight or for engine starting. The prime movers for large units are usually gas turbines, and for smaller units are gasoline engines or ram air turbines. The ram air turbine is classified as one kind of auxiliary power unit in this study. Electrical loads while the airplane is on the ground include galley services, cooling fans, recirculating fans, lighting, fuel pumps, instrumentation and ignition.

Gas turbine-generator APU: Usually the APU gas turbine uses the same fuel as the main engines. Unlike the main engines, these APU turbines run at constant speed. If the APU serves as a standby source of power in the event of failure of the main engines in flight, the turbine must be capable of being re-lit and operation at all altitudes. The gas turbine APU can provide a continuous supply of bleed air in combination with shaft power. Gas turbine APU can be placed advantageously anywhere in the airplane. The prime considerations are fire safety and noise.

The weight of a gas turbine varies appreciably with design requirements. For reference purpose, the weight of a gas turbine and gearbox vs shaft horsepower (sea level and static) is given in fig. 54. The specific fuel consumption of a gas turbine also depends on operational conditions. At sea level and static condition, the fuel rate is about .5 lb/hp-hr (.23 kg/hp-hr) for gas turbines in the 200- to 600-hp range. Fuel rate increases rapidly as the turbine rating decreases to below 100 hp. Fuel rate also increases with altitude, but decreases as the airplane speed increases.

The gas turbine APU has much larger energy density than storage batteries. For example, consider a 70-hp, 60-kVA (.8 pf) gas turbine APU.

Weight of 70-hp turbine and gearbox (fig. 59)	=	75 lb (34 kg)
Weight of 60-kVA generator	=	60 lb (27 kg)
Assume the total installed weight	=	155 lb (70 kg)

(The weight of the starting air bottle, battery charger, and control panels are all ignored in this comparison.) The fuel rate of the turbine is about .7 lb/hp-hr (.32 kg/hp-hr). For a 1/2-hr operation, the fuel weight is 35 lb (16 kg).

APU and fuel weight	190 lb (86.4 kg)
Emergency capability	48 kW for 1/2 hr

The same weight of Ni-Cd battery has energy storage of approximately:

$$\frac{190 \text{ lb} \times 13 \text{ W-hr/lb}}{1000} = 2.47 \text{ kW-hr}$$

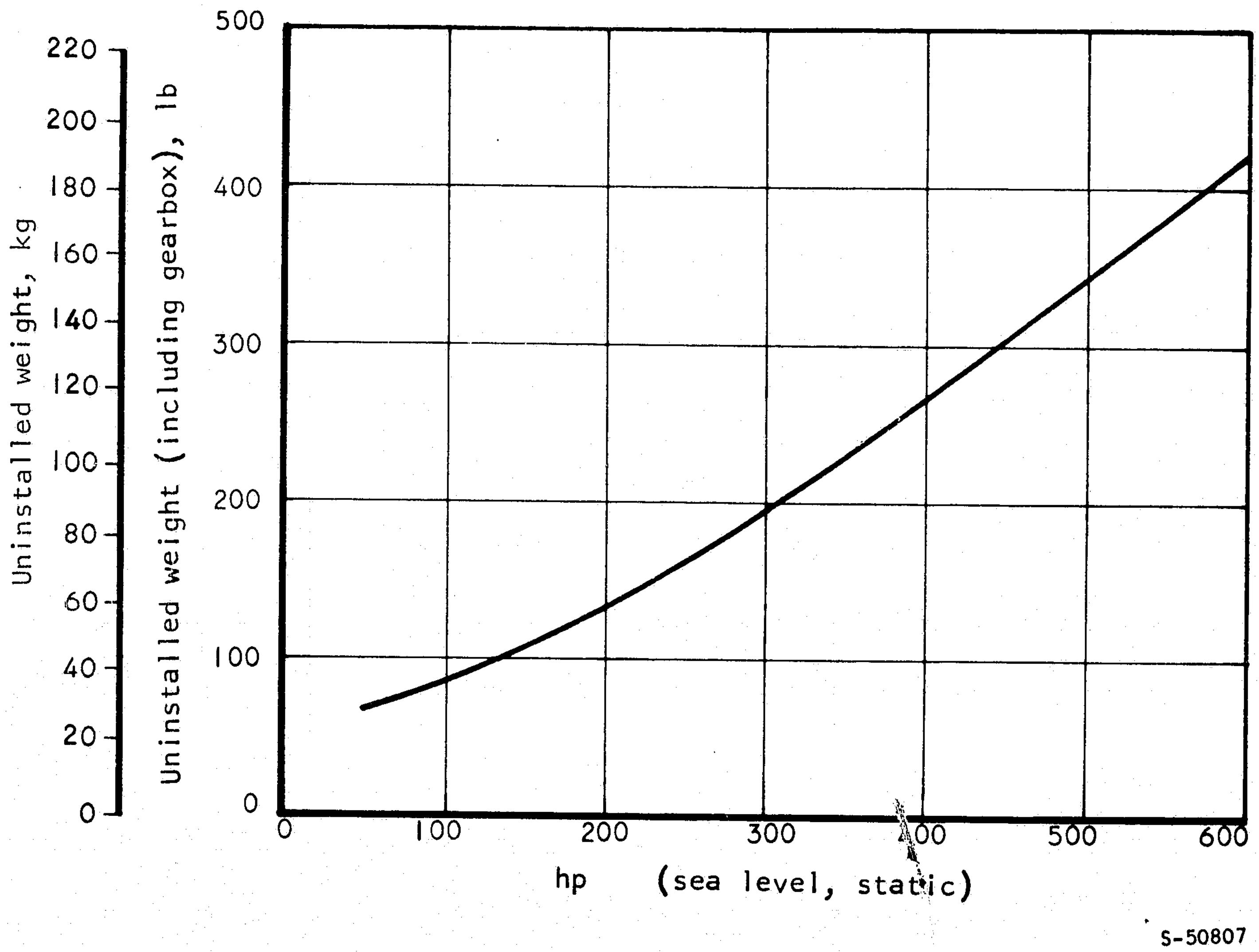


Figure 54. Approximate Weight of APU Gas Turbine with Gearbox

Thus the gas turbine APU in this given condition is a much better source of emergency power. However, if the emergency power requirement is low and the time duration for emergency power is short, the battery will become competitive with the gas turbine APU as an emergency power source.

Ram air turbine: The ram air turbine is another independent source of power for electrical, hydraulic, and shaft power for aircraft. These units generally mount in the fuselage of an aircraft and are flipped into the airstream when emergency power is required. If desired, these units can be retracted after use. Present ram air turbines can be used for flight envelopes ranging from as low as 80 knots to Mach 1.4 at sea level and up to Mach 2.4 at 60,000 ft (18,300 m) altitude. The speed of the turbine is controlled by a governor, presently at ±2 percent.

The ram air turbine may have various numbers of blades. In general, eight to ten blade units are best for hydraulic power systems where high starting torque and quick acceleration under load are required. These turbines develop high power at low rotational speeds. Two and four blade units are particularly suited to electrical power systems where a close speed control is required and high speed (12,000 rpm) generators can be used. These units are outstanding where light weight, low drag, and low costs are important requirements. Five-bladed units can fulfill the same requirements as two- and ten-bladed units. They have good starting torques and are designed for rotational speeds up to 9000 rpm.

The weight of ram air turbine varies with design condition. If the turbine supplies only electric power, the specific weight of turbine and generator combination is approximately 2 lb/kVA (.9 kg/kVA) for a 50-kVA unit to approximately 6 lb/kVA (2.7 kg/kVA) for a 5-kVA unit, at about 150 knots (77 m/sec) indicated airspeed and 12,000 rpm. The approximate diameter across the tips of the turbine blade is given in fig. 55.

Consider another example of providing 10 kVA (.8 pf) of emergency power for 10-min duration. A comparison of the weights of gas turbine APU, battery, and ram air turbine is given below.

(1) Gas turbine APU: 12.5-hp turbine, 10-kVA (8-kW) generator

Turbine and gearbox	60 lb (27 kg)
Generator	20 lb (9 kg)
Approximate combined installed weight	95 lb (43 kg)
Fuel for 10-min operation	5 lb (2.3 kg)
Total	100 lb (45 kg)

(2) Nickel-cadmium battery

$$\text{Energy required} = 8000 \times \frac{1}{6} = 1334 \text{ W-hr}$$

$$\text{Wt of battery} = \frac{1334}{13} = 103 \text{ lb } (47 \text{ kg})$$

(3) Ram air turbine

Estimated weight of turbine generator = 45 lb (20.5 kg)
(10 kVA, 12,000 rpm, 150 kias (77 m/sec)

It can be seen that at low power level, the ram air turbine APU has higher energy density than the nickel-cadmium battery and the gas turbine APU.

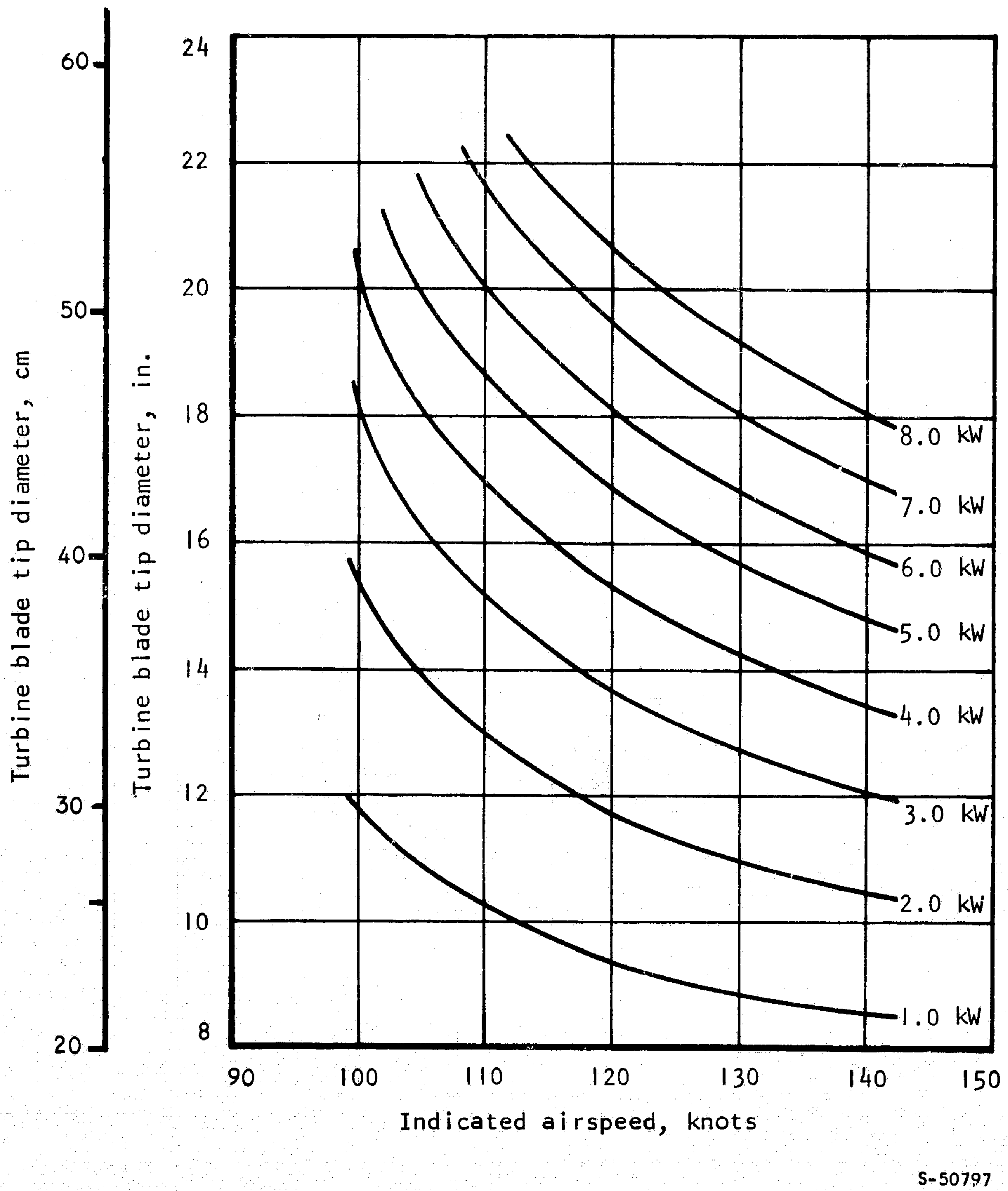


Figure 55. Turbine Blade Tip Diameters

RELIABILITY

Introduction

As applied to aircraft electrical power systems, the term "reliability" is employed to indicate the probability of not experiencing a component, channel, subsystem, or system failure, while the term "safety" is employed to indicate the implications of failure. While it is recognized that individual components must be made inherently as reliable as feasible to minimize required maintenance actions, it is mandatory that any system, to be suitable, be designed to tolerate both component failure and channel failure, since they are certain to occur, even if infrequently. Thus, a primary system objective is established, which shall be identified as a "failsafe criterion." (Failsafe means in this report either (1) the system has a backup which continues to function if the electric supply fails, or (2) the failure does not produce secondary damage.)

A corollary of the primary failsafe objective, and one with strong economic implications, is termed "dispatchability." This term is defined in current airline practice as follows: no single component failure (or channel failure) can be permitted to preclude a legal revenue dispatch of a commercial transport aircraft. This second objective should not apply automatically to small commercial or military aircraft, but its applicability should be determined by the FAA for the categories of smaller subsonic commercial aircraft.

These two objectives imply some degree of redundancy. Since increases in redundancy are accompanied by increases in initial cost, corrective maintenance frequency (the incidence of component, not system, failure determines the incidence of maintenance frequency), installed weight, complexity of control, etc., it is clear that any future system need not, and should not, possess redundancy appreciably in excess of that needed to attain full conformance to the two objectives of failsafe and dispatchability. In-flight reliability, defined as the ability to continue to the scheduled destination after an in-flight failure is also important, but usually is achieved if the failsafe and dispatchability objectives are realized.

Power System Channel Capacity

Availability of channels or complete systems for dispatch upon their intended mission is a probability combining reliability with accessibility and line, but not shop or depot, maintainability. In spite of some weapon systems engineers' having evolved some very sophisticated approaches to certain facets of their specialized problems, availability or reliability with maintenance, simply is "up time" divided by real or calendar time. Up time is usually approximated by mean-time-between-failures (MTBF), and calendar time is the sum of up time plus scheduled and unscheduled down time. Usually, down time equals the mean time to repair (MTTR) a system or to restore the system to its normal, fully operational status. Thus, from the point of view of a using airline (or armed service), availability is the salient characteristic which combines many of the previously listed characteristics.

It is important to note that availability is dependent upon the ratio of MTBF/MTTR where the denominator is not the mean-time-to-repair individual components (line replaceable units) but is the mean time to restore the system to its fully operational state. Thus, MTTR as employed in this context must reflect access time. It should also be noted that availability as defined above is equal to the maximum utilization factor (U_{max}), and that scheduled, as distinct from unscheduled, maintenance can reduce this factor. However, there exists a strong trend against scheduled maintenance and towards "on condition" maintenance. Also, electrical generating systems, like other systems, tend to receive their scheduled maintenance only when the aircraft as a whole is scheduled for such maintenance.

The importance of availability cannot be exaggerated. This is true for the dual reason that not only is it costly to maintain equipment, but the delay or cancellation of flights means that the aircraft users require more aircraft to maintain their intended level of operational capacity. If a four-engined aircraft has four channels of power generation and distribution each with an availability of A_1 and an unavailability of U_1 (the probability complement), then the availability of two (or more) channels out of four would be the summation of the terms of second power and higher in the expansion $A_{2/4} = (A_1 + U_1)^4 = A_1^4 + 4A_1^3U_1 + 6A_1^2U_1^2 + 4A_1^1U_1^3 + U_1^4 = 1$. The symbol $A_{2/4}$ indicates acceptable availability when two out of four channels must be fully operational (without fault). For the case $A_{3/4}$ (for dispatch) only the third and higher powers of A_1 are summed. If in a four-engined aircraft each channel were to have only 25 percent of design capacity, the summation of terms in the expansion might quite possibly be limited to the one (A_1^4) term, indicating that all four must be operational for dispatch. If, at the other extreme, each channel possesses a a design capacity of 100 percent, all but the last (U_1^4) term might be summed to indicate the potential for continuing an already airborne flight to the scheduled destination with three failed channels. If, as is almost certainly the only prudent approach, the on-board battery emergency supply is considered to be no more than an ultimate back-up supply, then (N-1) channels should be required for safe FAA approved dispatch.

At present, there is no set standard for generating capacity. From the above reasoning, it is clear that best practice lies between the two extremes cited and, in fact, lies between channel design capacities of 33 and 50 percent for four-engined aircraft. The logic employed for such problems can be somewhat different in emphasis between subsonic and supersonic aircraft. The logic employed in studies the past six years for cabin pressurization, refrigeration, and ventilation (usually referred to as the environmental control subsystem) for both the British-French SUD Aviation CONCORDE and the U.S. Boeing SST has been that takeoff shall be at 100 percent design capacity with one channel inoperative. The final design decision seems to rest upon the answer to this question: After a dispatch with one channel inoperative, and following an in-flight loss of a second channel, what is the minimum capacity required for

a safe continuation of the flight without serious change in flight plan or serious discomfort to the passengers or, more importantly, to the flight crew. The answer has usually laid between 67 and 100 percent of design capacity.

It is evident that certain design problems discussed herein are oriented to four-engined aircraft, different solutions are required for three- and two-engined aircraft. It is noteworthy that certain aerodynamic problems also relate to the number of engines, and it has been recognized for sometime that twin-engined commercial aircraft pose unique problems of design simply because an engine loss (on takeoff or climbout) constitutes a 50-percent loss of power.

Reliability of Electrical System and Components in Existing Aircraft

Many airlines have kept good maintenance and service records on aircraft components. Typical statistical data on component failure and repair are shown in table 17. This table, taken from Eastern Airlines Fleet Reliability Report, January 1969, shows the premature removal and failure rates for the top 40 components, ranked by failure rate, during the month of January 1969. Many of the electronic and control components have only several hundred hours of MTBF; therefore, they are the most unreliable. Electrical power components such as CSD, generator control panel, and ac generator are among the top 40 troublesome non-electronic components. The typical MTBF's of CSD transmission, generator control panel, and ac generator are 2700, 2900, and 4300 hr, respectively. A combination of these three components resulted in an MTBF of only about 1000 hr. Reliability improvement is needed in the present aircraft generation subsystem. Since the MTBF's of these three generation components are in the same orders of magnitude, reliability improvement of the generation subsystem can be obtained most effectively by improving the reliabilities of all three components.

A similar tabulation of unscheduled removal and failure rates of electric power generation components is shown in table 18. This table shows that the removal rate and failure rate of a certain component vary widely from month to month. To obtain typical failure data, average value from a large number of components or from many months' period should be used.

Delays and cancellations due to mechanical irregularities for various airplane fleets are shown in tables 19 through 23. These tables show that powerplant, landing gear, aircraft structure, and navigation system cause most of the delays and cancellations. Electric and hydraulic power systems have about the same reliability and are less troublesome than the power plant, landing gears, etc.

TABLE 17

TOP 40 COMPONENTS, RANKED BY FAILURE RATE, DURING JANUARY 1969*

Component	Type aircraft and no. of units in service	No. of sched. remov.	No. of prem. remov.	PRR** 1000 Unit hours	FR*** 1000 Unit hours
Electronic Components:					
Panel, auto-pilot control	DC9(84)	0	74	4.39	3.26
Vertical gyro	B720(30),L188(74)	0	51	3.09	2.12
Receiver, VHF comm.	All DC8(99),B720(45),L188(74)	0	94	2.98	2.03
Transceiver, VHF comm.	B727(225),DC9(252),DC8-61(48)	0	252	2.04	2.00
Computer, A/P air data	DC9(84),DC8-61(16)A/C #781	0	83	3.92	1.89
Receiver, VHF nav.	DC8-21/51(34),B720(30),L188(74)	0	79	3.38	1.80
Indicator, turn and slip	DC9(168)	0	54	1.60	1.30
Recorder	All except CV440(242)	0	76	1.44	1.21
Transceiver, DME	DC9(168),B727(150,B720(30), All DC8(66),L188(37)	0	186	1.76	1.14
Gyro, directional	DC9(168),B727(150),B720(30), All DC8(66)	0	144	1.50	1.00
Indicator, course deviation	B727(150),DC9(168),DC8-61(32)	0	103	1.26	.96
Transmitter, radar	B727(75),DC9)84),DC8-61(16)	0	122	2.99	.88
Reproducer, tape	All A/C except CV440(242)	0	60	1.23	.88
Indicator, horizon	DC9(168),B727(150),DC8-61 & #781(34)	1	87	1.06	.73

*Eastern Airline Fleet Reliability Report, January 1969
**PRR - Premature removal rate
***FR - Failure rate

TABLE 17.--Continued

Component	Type aircraft and no. of units in service	No.of sched. remov.	No.of prem. remov.	PRR* 1000 Unit hours	FR** 1000 Unit hours
Computer, pitch	DC9(84)	0	66	3.91	.59
Receiver, ADF	B727(150),DC9(168),DC8-61(32)	0	61	.75	.50
Transponder, ATC	DC9(168),B727(150),All DC8(66), B720(30),L188(74)	0	119	1.13	.38
Gyro, vertical	DC9(252),B727(105),DC8-21/51(35) DC8-61(48)	2	64	.91	.38
Transceiver	DC9(84),B727(75),DC8-61(16), A/C #779 & 781	0	44	1.06	.36
Receiver	DC9(252),B727(225),DC8-61 & #781(51)	0	115	.93	.29
Components Other Than Electronic:					
Control, heat and Cab. Press.	L188(34)	0	24	5.86	3.42
Synchronizer, cab. press.	L188(34)	0	28	6.84	2.44
Controller, cabin temp.	L188(34)	0	43	10.50	2.20
Indicator, fuel qty. center M.	L188(136)	0	34	2.08	1.04
Clock	B727(150)	0	55	1.39	.88
Pump, hydraulic	B727(150)	7	37	.93	.81
Indicator, EPR	DC8-21(56)	0	29	2.27	.78
Clock	DC9(176),DC8-61(48),B727QC(75)	0	52	.61	.53
Indicator, T/M	L188(136)	0	27	1.65	.49

*PRR - Premature removal rate
**FR - Failure rate

TABLE 17.--Concluded

Component	Type aircraft and no. of units in service	No.of sched. remov.	No.of prem. remov.	PRR* 1000 Unit hours	FR** 1000 Unit hours
***Control panel, AC Gen.	DC9(252)	0	40	.79	.47
***Transmission, CSD	B727(225)	0	28	.47	.37
***Control panel, AC Gen.	DC9(252)	0	32	.63	.32
Indicator, fuel Flow	DC8-21/61(120)	0	35	1.20	.31
***Generator	B727(300),B720(60)	33	58	.60	.27
***Generator Control panel	B727(300)	0	31	.39	.27
Amplifier, vibration	DC8(33),B727(75),B720(15)	0	51	1.60	.25
Indicator, EPR	DC9(176)	0	33	1.96	.24
Indicator, aux. fuel qty.	DC9(176)	0	31	.92	.18
Power supply, fuel flow	DC9(164)	0	31	.92	.18
Indicator, turbo comp. rpm	DC8-21/61(60)	0	31	2.12	.14

*PRR - Premature removal rate
**FR - Failure rate
*** - Electrical power component

TABLE 18

UNSCHEDULED REMOVALS*

Part number description	Aircraft type	October 1968			November 1968			December 1968			January 1969			FR P/S
		UR	URR	FR	UR	URR	FR	UR	URR	FR	UR	URR	FR	
ELECTRIC POWER (ATA 24)														
21-GE-0002 Ac generator	DC8-51 & #781	2	.15	.08	5	.37	.07	3	.18	.06	0	0	0	
21-GE-0003 Ac generator	DC8-51/-55	3	1.98	-	4	2.22	.56	1	.38	-	4	1.58	.79	
21-GE-0013 Ac generator	CV-440	1	.24	.24	1	.27	-	0			0			
21-GE-0016 Ac generator	DC-9	22	.53	.36	11	.25	.16	6	.12	.08	10	.20	.20	.25
21-GE-0017 Ac generator	L-188	11	.61	.28	11	.62	.34	18	1.03	.57	6	.37	.18	.35
21-GE-0018 Ac generator	B-720/B-727	21	.24	.10	26	.29	.17	47	.50	.32	58	.60	.27	.15
21-GE-0020 Ac generator	DC8-21	1	.08	-	5	.38	.08	1	.07	-	4	.33	.33	.15
21-TR-0101 CSD transmission	DC8-21/ B-720	8	.30	.15	12	.43	.18	17	.55	.23	17	.59	.24	.40
21-TR-0103 CSD transmission	DC8-51/55 (-51,2 AC)	1	.66	-	3	1.66	.55	2	1.15	-	4	2.36	.59	
21-TR-0105 CSD transmission	B-727	27	.49	.46	31	.56	.45	39	.67	.60	28	.47	.37	

*Eastern Airline Fleet Reliability Report, January 1969

LEGEND: UR = Unscheduled removals URR = Unscheduled removal rate FR = Failure rate
P/S = Performance standard (3 month average) 00 = Above P/S, one month

TABLE 19

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (B-720) JANUARY 1969*

ATA code	Number of events					Percent departures delayed or cancelled		
	Delays	Min	Can-celled	Other	Total	Report Month	3-Mo avg	Perf std
32 Landing Gear	22	1520	2	4	28	1.12	.64	.70
71-80 Power Plant	8	364	-	10	18	.42	.42	
71-80 Aircraft	5	190	1	20	26	.38	.38	
29 Hydraulic Power	4	476	3	-	7	.33	.31	.30
34 Navigation	5	120	1	9	14	.28	.24	.20
21 Air Conditioning	3	119	-	27	30	.14	.18	.12
26 Fire Protection	1	67	1	-	2	.09	.13	.10
27 Flight Controls	2	249	-	-	2	.09	.11	.20
33 Lights	1	34	-	5	6	.05	.10	.10
24 Electrical Power	1	5	-	22	23	.05	.10	.11
28 Fuel	-	-	1	12	13	.05	.06	.15
30 Ice & Rain Protection	1	67	-	2	3	.05	.05	.03
38 Water & Waste	1	49	-	-	1	.05	.03	.02
22 Auto Pilot	1	99	-	11	12	.05	.03	.07
23 Communications	1	80	-	5	6	.05	.03	.07
57 Wings	1	39	-	-	1	.05	.03	N.A.
52 Doors	-	-	-	1	1	-	.03	.15
05 Miscellaneous	1	62	-	-	1	N.A.	N.A.	N.A.
TOTALS	59	3677	9	127	195			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 20

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (DC8-21-51) JANUARY 1969*

ATA code	Number of events					Percent departures delayed or cancelled		
	Delays	Min	Can-celled	Other	Total	Report month	3-Mo avg	Perf std
32 Landing Gear	10	1116	-	-	10	.61	.70	.70
71-80 Power Plant	11	1155	1	13	25	.70	.65	
71-80 Aircraft	3	75	-	25	28	.55	.55	
27 Flight Controls	5	700	2	8	15	.43	.47	.35
34 Navigation	6	295	-	11	17	.36	.41	.35
52 Doors	-	-	2	5	7	.12	.21	.15
28 Fuel	2	77	-	28	30	.12	.19	.16
24 Electrical Power	-	-	1	8	9	.06	.17	.18
29 Hydraulic Power	2	89	-	1	3	.12	.13	.30
33 Lights	2	70	-	-	2	.12	.13	.10
35 Oxygen	1	12	-	1	2	.06	.11	.08
38 Water & Waste	1	22	-	-	1	.06	.06	.05
56 Windows	1	275	-	-	1	.06	.06	N.A.
36 Pneumatic	1	14	-	2	3	.06	.06	.12
54 Nacelles/Pylons	2	305	-	1	3	.12	.04	N.A.
26 Fire Protection	2	233	-	-	2	.12	.04	.10
21 Air Conditioning	-	-	-	24	24	-	.02	.12
22 Auto Pilot	-	-	-	10	10	-	-	.07
05 Miscellaneous	4	122	-	7	11	N.A.	N.A.	N.A.
TOTALS	53	4560	6	144	203			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 21

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (DC8-61) JANUARY 1969*

ATA code	Number of events					Percent departures delayed or cancelled		
	Delays	Min	Can-celled	Other	Total	Report month	3-Mo avg	Perf std
71-80 Power Plant	3	481	-	10	13	.27	.65	
71-80 Aircraft	5	289	-	12	17	.55	.55	
32 Landing Gear	4	150	1	-	5	.34	.39	.70
34 Navigation	4	231	-	17	21	.27	.34	.35
29 Hydraulic Power	5	445	-	2	7	.34	.22	.30
52 Doors	5	250	-	10	15	.34	.20	.15
24 Electrical Power	3	120	-	6	9	.20	.17	.18
27 Flight Controls	2	207	-	2	4	.13	.15	.35
28 Fuel	2	46	-	32	34	.13	.10	.16
21 Air Conditioning	2	99	-	33	35	.13	.10	.12
25 Equipment & Furnishings	-	-	-	1	1	-	.10	.05
33 Lights	-	-	-	6	6	-	.10	.10
26 Fire Protection	1	90	1	-	2	.13	.07	.10
30 Ice & Rain Protection	1	182	-	3	4	.07	.05	.08
36 Pneumatic	-	-	-	7	7	-	.05	.12
23 Communications	-	-	-	5	5	-	.02	.12
38 Water & Waste	-	-	-	1	1	-	.02	.05
22 Auto Pilot	-	-	-	7	7	-	-	.07
31 Instruments	-	-	-	2	2	-	-	.04
05 Miscellaneous	1	64	-	2	3	N.A.	N.A.	N.A.
TOTALS	38	2654	2	158	198			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 22

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (L-188) JANUARY 1969*

ATA code	Number of Events					Percent departures delayed or cancelled		
	Delays	Min	Can-celled	Other	Total	Report month	3-Mo avg	Perf std
71-80 Power Plant	14	1118	1	27	42	.35	.58	
71-80 Aircraft	6	109	1	7	14	.48	.48	
32 Landing Gear	15	709	2	-	17	.35	.28	.25
34 Navigation	8	249	-	34	42	.17	.16	.20
24 Electrical Power	4	179	-	8	12	.08	.12	.13
21 Air Conditioning	7	285	1	31	39	.17	.09	.18
33 Lights	8	145	-	4	12	.17	.08	.07
29 Hydraulic Power	4	194	-	5	9	.08	.07	.08
56 Windows	1	157	2	-	3	.06	.05	N.A.
23 Communications	3	20	-	4	7	.06	.05	.10
30 Ice & Rain Protection	2	218	-	6	8	.04	.05	.05
28 Fuel	2	119	-	19	21	.04	.03	.07
25 Equipment &	1	12	-	3	4	.02	.03	.02
27 Flight Controls	1	77	-	-	1	.02	.03	.10
52 Doors	1	42	-	13	14	.02	.03	.125
36 Pneumatic	2	102	-	1	3	.04	.02	.025
38 Water & Waste	-	-	1	-	1	.02	.01	.025
22 Auto Pilot	-	-	-	9	9	-	-	.015
05 Miscellaneous	2	14	-	2	4	N.A.	N.A.	N.A.
TOTALS	81	3749	8	173	262			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 23

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (CV-440) JANUARY 1969*

ATA code	Number of events					Percent departures delayed or cancelled		
	Delays	Min	Can-celled	Other	Total	Report month	3-Mo avg	Perf std
71-82 Power Plant	14	1364	10	1	25	1.30	1.25	
71-82 Aircraft	2	239	1	6	9	1.10	1.10	
32 Landing Gear	8	382	-	-	8	.35	.26	.30
26 Fire Protection	7	237	-	-	7	.31	.23	.15
24 Electrical Power	4	57	2	-	6	.26	.11	.175
33 Lights	4	91	-	2	6	.18	.11	.10
34 Navigation	2	73	-	1	3	.09	.11	.12
21 Air Conditioning	1	86	-	1	2	.04	.09	.10
23 Communications	2	19	-	-	2	.09	.08	.10
29 Hydraulic Power	1	86	1	-	2	.09	.05	.05
56 Windows	-	-	1	1	2	.04	.05	N.A.
52 Doors	1	139	-	1	2	.04	.04	.125
30 Ice & Rain Protection	-	-	-	1	1	-	.03	.03
27 Flight Controls	1	60	-	-	1	.04	.01	.03
35 Oxygen	1	40	-	-	1	.04	.01	.01
05 Miscellaneous	1	13	-	1	2	N.A.	N.A.	N.A.
TOTALS	49	2886	15	15	79			

*Eastern Airline Fleet Reliability Report, January 1969

HEAT TRANSFER TECHNIQUES

Introduction

The problem of cooling electric and electronic equipment in aircraft, in general, has become increasingly difficult due to:

(1) Requirements for operation at increasingly high altitudes and flight speeds

(2) The size reduction of equipment to fit allowable space envelopes

(3) An increase in the power ratings of components

(4) The desire to increase the present service life components by operating at lower temperatures

To meet the specified performance and reliability criteria of an electric or electronic component, cooling is normally necessary to maintain the component temperature at or below the manufacturer's maximum rated temperature for at least 90 percent of the life of the unit.

Reliability requirements are usually expressed as a mean-time-between failure (MTBF), the reciprocal of which is defined as the failure rate. From failure-rate curves, such as shown in figs. 56 and 57, the allowable component ambient temperature can be determined for a required MTBF for a specific component.

The recommended component ambient temperatures as specified in MIL-E-5400 are reproduced in table 24. For continuous operations, temperatures of 55°, 71°, 95°, and 125°C are not to be exceeded for Classes 1, 2, 3, and 4 environments, respectively, under the worst heat dissipation conditions.

Component Cooling Methods

The cooling methods now being used in aircraft electrical equipment are grouped subjectively into seven categories listed below. These methods are generally special cases or combinations of the basic heat transfer mechanisms, i.e., conduction, convection, and radiation.

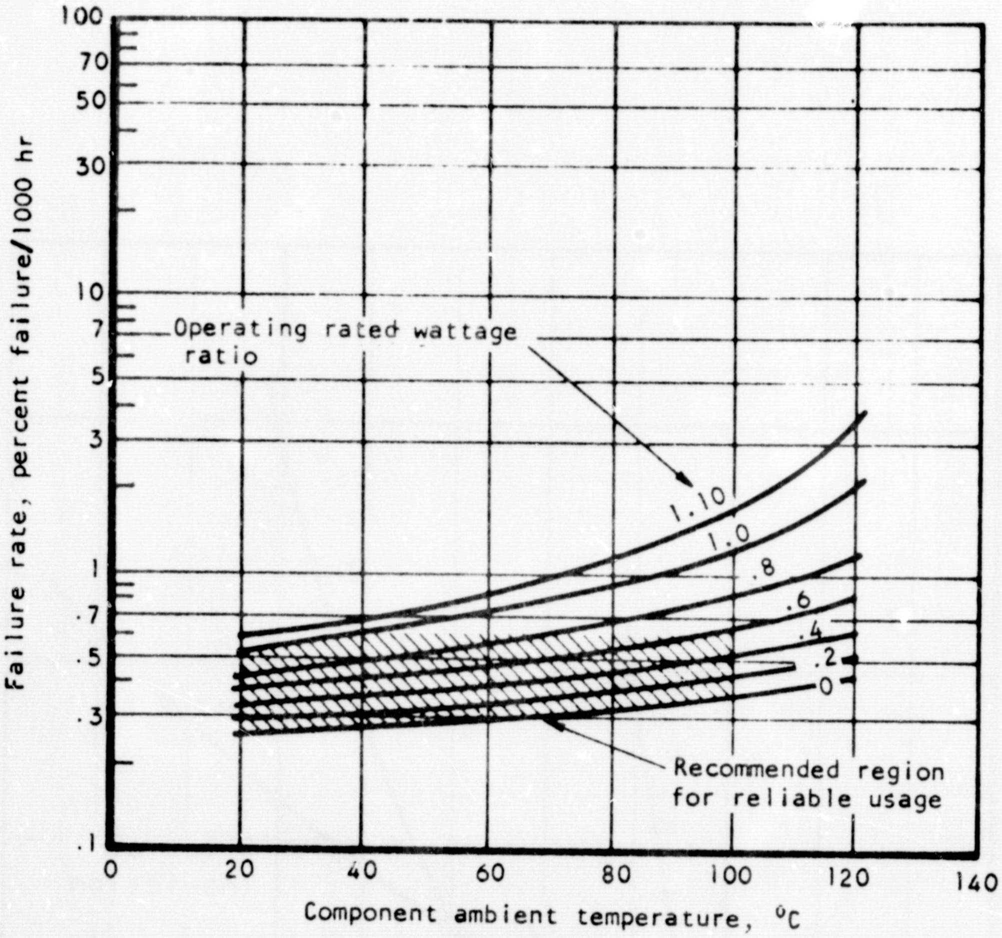
(1) Self-cooling

(2) Conduction or heat-shunt cooling

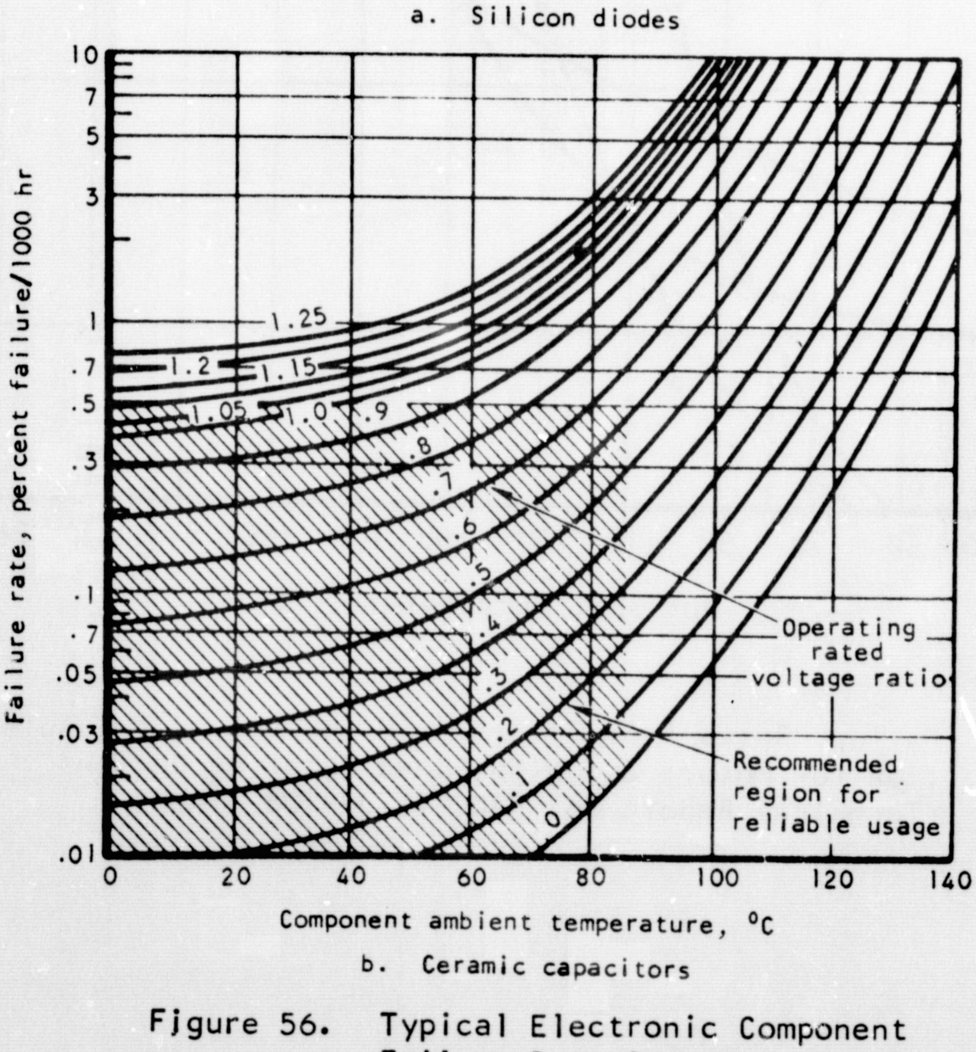
(3) Forced gas cooling

(4) Forced liquid cooling

(5) Thermal attenuation or thermal damping



a. Silicon diodes



b. Ceramic capacitors

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Figure 56. Typical Electronic Component Failure Rate Curves

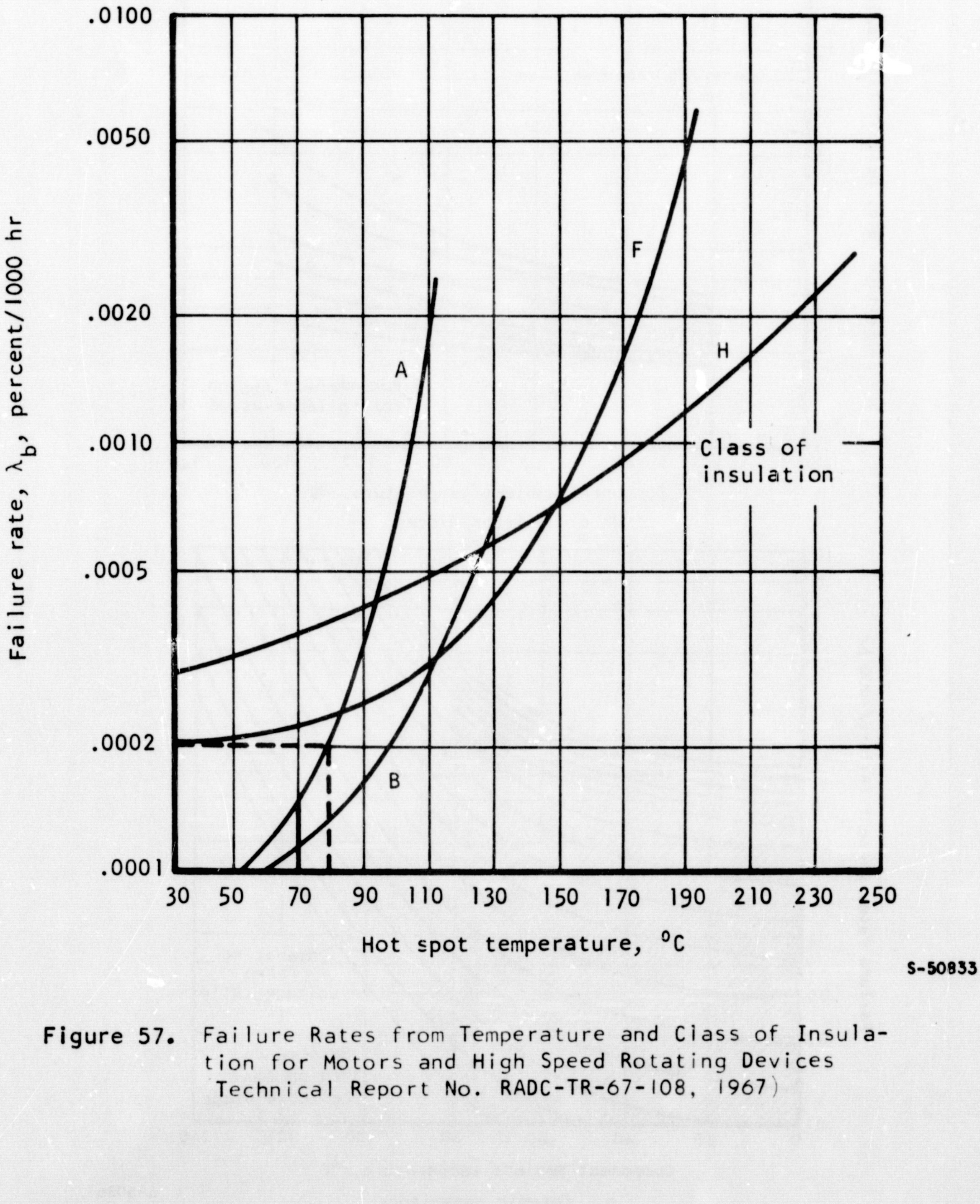

Figure 57. Failure Rates from Temperature and Class of Insulation for Motors and High Speed Rotating Devices Technical Report No. RADC-TR-67-108, 1967)

TABLE 24

RECOMMENDED COMPONENT AMBIENT TEMPERATURES (MIL-E-5400)

Environment	Ambient temperature limit, °C			Heat flux, W/sq in.	Applications
	Preferable continuous operation	Maximum continuous operation	Maximum intermittent operation		
Class 1	25	55	71	.1 to .5 (.016 to .078 W/cm^2)	Integrated circuits, capacitors, resistors, and power transistors
Class 2	40	71	95	0.3 (.047 W/cm^2)	Integrated circuits, transistors, resistors, and high-temperature capacitors
Class 3	70	95	125	----	Motors, generators, transformer, inductors, diodes
Class 4	90	125	150	----	High-temperature components

(6) Thermoelectric coolers and heat pumps

(7) High-temperature heat exchangers

These methods of cooling electrical system components and cooling systems now in use on aircraft are briefly discussed in Appendix D. The relationships between the basic cooling methods and the heat dissipation capabilities as normally encountered in electronic equipment are shown in fig. 58. The method of boiling heat transfer is also included to obtain more combinations of internal and external cooling modes for comparison.

Table 25 shows the approximate cooling capabilities of several methods that have been used successfuly for electrical rotating machines. The numbers indicated are somewhat higher than the corresponding ones in electronic cooling of fig. 58. This is because in rotating machines higher temperature differences are normally feasible for cooling purposes.

The factors that may influence the cooling capability of a particular application are so many that the heat dissipation capabilities shown in fig. 58 and table 25 may only be regarded as a guide.

TABLE 25

APPROXIMATE COOLING CAPABILITIES OF DIFFERENT METHODS FOR ELECTRIC ROTATING MACHINES

Cooling method	Approximate cooling capability
Self-cooling without heat sinks	3 W/cu in. (.18 W/cm^3)
Self-cooling with heat sinks	5 W/cu in. (.31 W/cm^3)
Forced gas convection	5 to 9 W/cu in. (.31 to .55 W/cu cm^3)
Forced liquid convection	10 to 20 W/cu in. (.61 to 1.22 W/cm^3)
Expendables	50 or more W/cu in. (3.1 W/cm^3 or more)

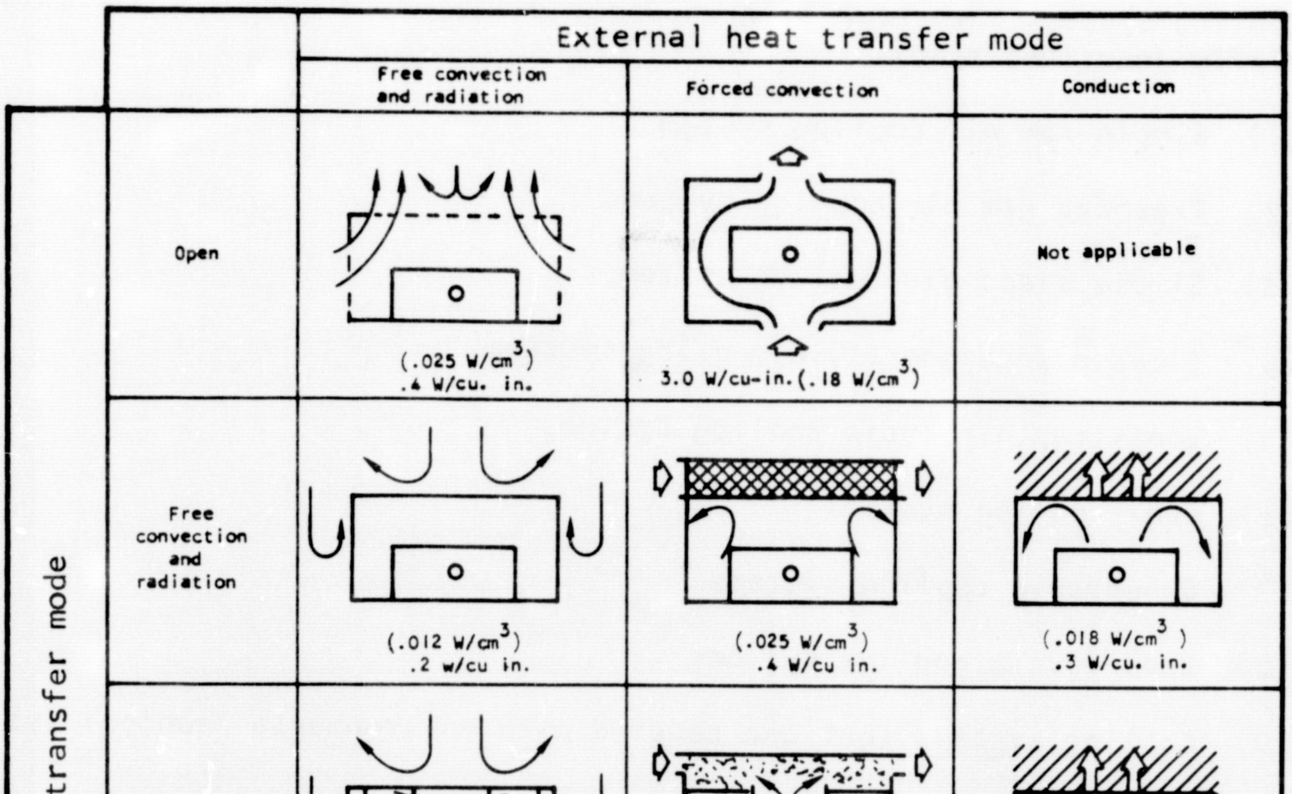

Figure 58. Internal/External Heat Transfer Modes and Heat Dissipation Capability

Cooling Systems

Cooling systems that have been used for different aircraft applications are briefly described below.

(1) Simple ram air cooling system

(2) Expanded ram air cooling systems

(3) Simple bleed air cooling system

(4) Bleed air regenerative cooling systems

(5) Bootstrap air cycle cooling systems

(6) Fuel cooling systems

(7) Expendable cooling systems

(8) Vapor-cycle cooling systems

(9) Cooling systems that are coupled with environmental control systems

This list consists of two ram air systems, three bleed air systems, and four others. Descriptions of these systems are given in Appendix D of this report. Variations and modifications of these basic systems are possible but are not described. The system control that is an integral part of the basic system is not discussed in detail.

It is highly improbable that one single cooling system will be optimum for all flight conditions. A combination of several systems and cooling techniques, e.g., bootstrap air cycle system for environmental control, liquid loop to fuel for electrical cooling plus cryogenic expendables for a few components and thermoelectric spot-cooling for some microcircuits, may turn out to be the lightest weight system for a particular aircraft for a specific mission.

A computer program (H-0610) may prove to be very useful in the optimization procedure. A sample summary output for a 160-kW thermal load and 5.5 engine compression ratio from this program is shown in fig. 59. It is a map showing the minimum weight system for each combination of Mach number and altitude. For example, fig. 59 shows that at Mach number (abscissa) of 1.5 and altitude (ordinate) of 60,000 ft (18,200 m), the minimum weight system (indicated by a circle) would be expendable evaporants (expendable system abbreviated as VAP). The approximate mission profile for an F-111 aircraft is plotted overlapping the computer outputs (the dotted curve in the figure), showing the best solution to every flight condition for the complete mission. As it is indicated in the output, the minimum weight cooling system during takeoff would be a simple bleed air system (SIM). For climb, cruise, and descent, either an expendable system (VAP) alone or an integration of simple ram air system with a liquid transport loop (LIQ), expendable system (VAP), and bleed air regenerative system (REG) could be used.

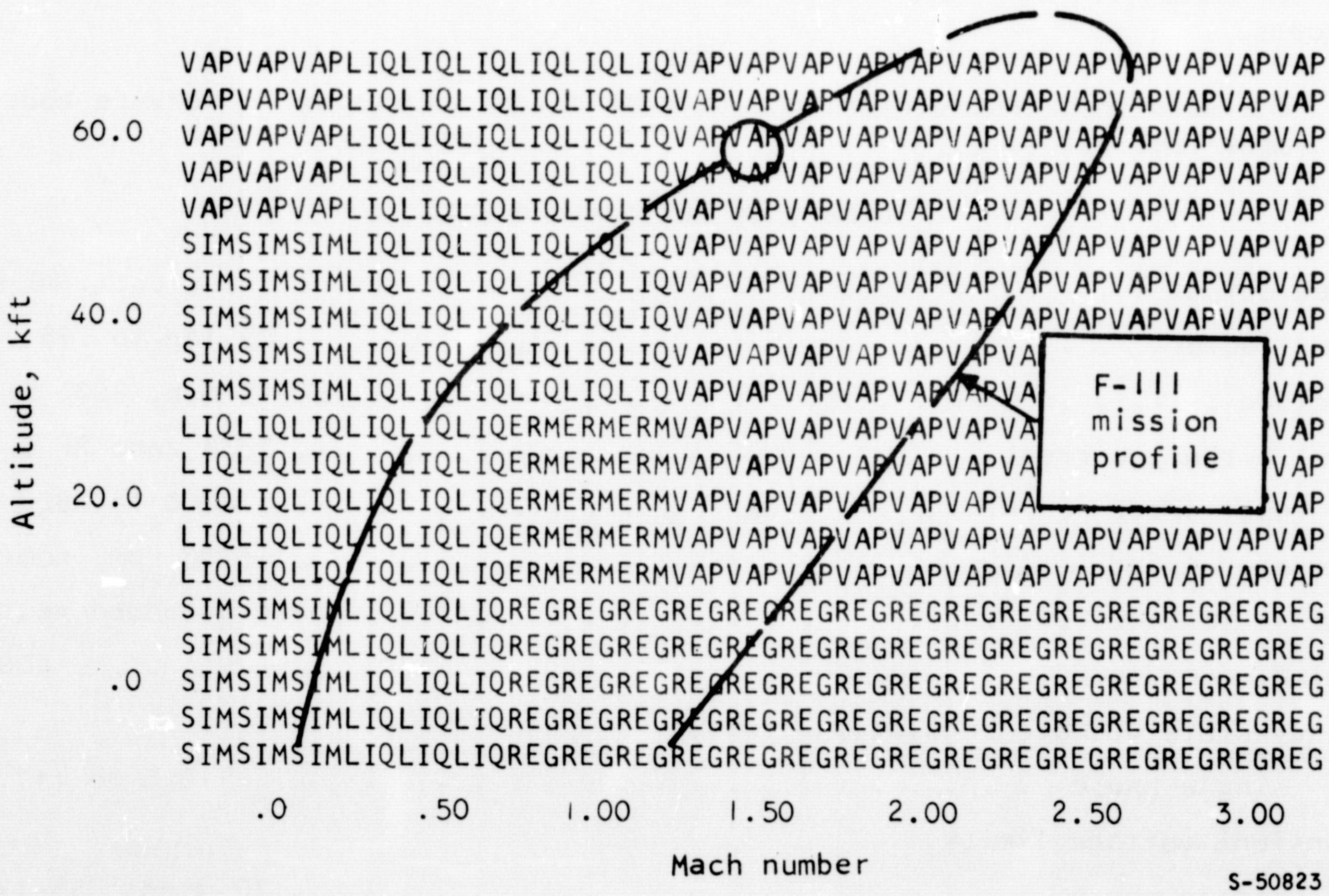

Figure 59. Output Data Plot of Minimum Weight Systems

CAPABILITY CONSTRAINTS OF EXISTING SYSTEM

Contemporary aircraft electrical systems generally conform to the specified requirements. These requirements were established more than a decade ago. Since then technology has advanced and the requirements of larger and faster airplanes have brought about the need for changing existing specifications. It may be more appropriate to speak of outdated specifications than of capability constraints. To change these specifications it first appears necessary to progress with technological advances and to continuously evaluate the feasibility of new concepts.

The more important performance data of a typical system in service today are summarized below.

Summary of Performance Data

Rated output	40 kVA
Power factor	.75 lag to .95 lead
Overload	150% 5 min, 200% 5 sec
Short circuit current	333A zero PF 5 sec
CSD input speed	4000 to 7800 rpm
Generator speed	6000 rpm, nominal
Nominal voltage	115/208 3-phase grounded neutral
Nominal frequency	400 Hz
Steady-state voltage regulation limit, single phase	110.5 to 117.5 V
Transient voltage limits, single phase	20 msec: 55-170 V 100 msec: 65-160 V 500 msec: 95-135 V
Phase displacement	$120^0 \pm 1.5^0$
Phase unbalance	3V max between phases
Waveform	Crest factor 1.41 ±.1
Harmonic content	5% total, 3% each individual
Circulating reactive current between systems (steady state)	1.9 kVAR
Circulating real current between systems (steady state)	4 kW

These data pertain to a generating system using a hydraulic constant speed drive with conventional control and protection. The system normally consists of two to four generators operating in parallel.

Aircraft air cooled ac generators and regenerators are specified in MIL-G-6099 (latest revision A). This specification was originated in 1950, revised in 1957 and again amended in 1958. Recently it has been the practice to utilize oil cooled generators with integral permanent magnet generators for system self-sufficiency. These generators exhibit overload characteristics which are presently not covered by government specifications. A typical performance characteristic of one of the newer generators is 1.5 per unit continuous output and 2 per unit for 15 sec at 0.75 pf with 150°C oil inlet temperature.

Power quality specification MIL-STD-704 was originated in 1955 and has been amended several times since, most recently in February 1968. In MIL-STD-704, utilization equipment are grouped into categories A, B and C. The line voltage drop from the point of regulation to the utilization equipment is limited to 2 Vac or .5 Vdc for category A; 4 Vac or 1 Vdc for category B; and 8 Vac or 2 Vdc for category C equipment which is required to operate intermittently only. The steady-state voltage limits from which the three categories of equipment are required to operate under different modes of electric system operation are listed in table I for ac and table II for dc in MIL-STD-704.

A few of the aircraft electrical equipment design constraints are discussed below.

Generation equipment.--At present, the MIL-G-6099 specification calls for a 3 per unit short circuit current for the ac generator to ensure fast tripping of the circuit breaker. This requirement imposes weight penalty to the generator due to the low synchronous impedance and high excitation ceiling. Use of solid-state circuit breakers can remove this design limitation.

In a 400-Hz system, the maximum generator speed is limited to either 12,000 or 24,000 rpm. (A 24,000 rpm generator weighs about the same or slightly heavier than a 12,000-rpm machine because the 2-pole configuration is not as favorable as the 4-pole configuration.) If a dc or higher frequency system is used, however, the operating speed of the generator can be increased and some weight saving can be accomplished.

In the present brushless ac generator, the output of the voltage regulator goes to the exciter field and then corrects the generator voltage. The rate of response is relatively slow. Slow response causes large voltage transient which in turn penalizes some of the utilization equipment. Some new excitation schemes can increase the response rate to some extent. The VSCF system will reduce this transient response time drastically.

Distribution equipment.--One of the drawbacks of ac transmission and distribution is the reactance line drop with the associated induced eddy current losses. For large aircraft, a high dc voltage distribution system could reduce the weight of power wiring by a substantial amount.

Low frequency ac (i.e., 10 to 500 Hz) is the most dangerous electric current for human beings. Dc or high-frequency ac (kilo-Hertz) power will reduce the hazard of electric shock to operation personnel.

Modern aircraft require an alarmingly large amount of signal wiring. High strength small gage wires or signal multiplexing becomes a necessity to keep the wiring weight to a reasonable level.

Connectors at present give the most trouble to the aircraft electrical power system. Their improvement deserves imminent attention.

Conventional electromagnetic circuit breakers and power control devices are slow in operation. Solid-state devices are much faster but have other drawbacks. For instance, solid-state circuit breakers have high forward drop, and have no complete isolation between the power and signal circuits. Also, solid-state control devices are sensitive to electric noises and power transients. A tradeoff between the pros and cons should be examined before a selection is made.

When the titanium frame is used for aircraft, the present practice of using aircraft skin as electrical return will no longer be practical. The penetration of electrical wiring through the aircraft frame partitions weakens the airframe structure, which in turn increases the weight of the aircraft structure. A review of the wire routing may be beneficial to aircraft weight saving.

Utilization equipment.--It is understood that the power supply cannot satisfy the many different requirements of various utilization equipment, and that it is not practical to provide the most precise and closest tolerance power to the entire system when only a small fraction of the load demands this high level quality power. These equipment then would have to be supplied via a power conditioning device and/or be furnished with ultimate reliable power for that case where power interruptions cannot be tolerated.

The influence by utilization on the characteristics of the power, which is linked to all equipment should be defined and be restricted to limit interaction to a minimum. For this the characteristic of the distribution system becomes important and parameters such as the surge impedance of various branches of the distribution system should be specified.

Some avionics suppliers expressed the opinion that if there is a higher quality power than presently available in aircraft, some saving in weight of avionics equipment can be achieved.

400 Hz operation limited the maximum motor rpm speed to 24,000 rpm. Present aircraft electric motors can be reduced in size and weight if higher operation speed (even with reduction gear to yield the same output speed) and advanced materials are used.

Brushless dc motors and power converters demand heavy EMI suppression equipment. Better means of EMI suppression would therefore be helpful.

System protection.--Existing systems provide a minimum number of essential protective functions to guard the generating system and its connected loads from harmful abnormal system conditions. Some harmful conditions occur more frequently than others. For instance, experience has shown that an open phase condition is more likely to occur than a short circuit on the synchronizing bus. Yet many systems do provide synchronizing bus protection and neglect to protect for the condition of an open phase. Indications and displays of equipment malfunctions are only of value to the flight crew, when corrective or further preventive action may be taken. The operation of the system should be made simpler by automated programming. Future advancements in protective and control schemes will include:

(1) More than one differential protection zone to include synchronizing bus and ground feed-ties.

(2) Better selectivity of system faults during parallel operation.

(3) Fully automated parallel connection via generator circuit breaker or bus tie breaker selection of faulted system and reclosure of BTB.

(4) Automated load shedding and current limiting.

SURVEY OF TECHNOLOGY IMPROVEMENT ACTIVITIES

Recently, many activities to improve aircraft electrical power systems have been initiated in various sectors of industry. Based on the literature and industry survey, a summary of these activities is presented below.

Power Generation

During the past few years, the conventional generating system has been improved while a new generating system was being developed. The results of this simultaneous activity are the integrated drive generator (IDG) and the variable speed constant frequency (VSCF) system.

Integrated drive generator.--Recent advancements in the development of a conventional generating system are demonstrated by the new IDG now undergoing flight test. This new drive is presently selected for Lockheed L-1011 aircraft. The package combines a hydraulic transmission and a generator built into an integral housing; the generator is oil splash cooled and turns at 12,000 rpm. A marked improvement is exhibited by the IDG package over older CSD systems which has made it increasingly more dificult for the all-electric VSCF system to gain recognition. A 60-kVA IDG presently weights 115 lb but is expected to weigh only 95 lb in the near future. The quoted reliability of this new drive is 12,000-hr MTBF.

VSCF system.--Since the advent of high-power silicon controlled rectifiers, constant frequency can be obtained through a static frequency changer convected to the variable frequency generator, and the SCD is eliminated. An advantage of the VSCF system is its potential high reliability because very few moving parts are involved. It is estimated that the eventual MTBF of the VSCF system will be several times longer and that the maintenance cost will be much lower than that of a corresponding CSCF system (refs. 13 and 14).

There are two main schemes for the VSCF power generation--dc link and cycloconverter. At the present time they are both heavier than the CSD method if the same conventional generators are used. If the generator speed could be increased, the weight of the VSCF system would be less than that of the CSD system; however, the speed of the generator has to be high (24,000 rpm or higher) before the weight of this system becomes lower. At such a high speed, stress problems are encountered. One attractive design solution is to use a solid rotor generator. But some stress problems are still unsolved.

The cycloconverter method is more attractive than the dc link method because of lower system weight and smaller volume. But the cycloconverter has an additional frequency interference problem and a more stringent requirement on the generator subtransient reactance. If the stress and the interference problems are solved, future aircraft will use cycloconverter VSCF systems.

4

If the operating frequency of the equipment remains at 400 Hz, the input frequency to the cycloconverter will be at least 1200 Hz (input to output frequency ratio should be at least 3 to 1). This high frequency power would have the following benefits:

(1) Reduce transformer size and weight

(2) Reduce filter weight and size for dc supply due to high ripple frequency

(3) Improve frequency regulation, frequency modulation, and transient performances

The following performance improvements demonstrated by the new VSCF cycloconverter system might be stipulated as required improvements for all future systems:

(1) Switching transients confined to extremely close limits

(2) Drastically reduced transient response time

(3) Better phase balance by individual voltage regulation of phases (improves system characteristic during unbalanced loading and faults)

(4) Extremely precise frequency regulation

(5) Elimination of frequency transients

(6) Almost complete elimination of circulating currents between paralleled systems because closer tolerance power is obtainable; redundant derating channel capacity for parallel operation.

The VSCF cycloconverter system may be programmed to supply various frequency levels including dc. In case of an engine failure the system may be utilized to supply emergency dc power when driven by a windmilling engine. The chief disadvantage of any VSCF scheme is that it is considerably heavier than the IDG unit. Its reliability is not yet proven, however, the VSCF has great potential including that of superior maintainability.

A VSCF cycloconverter system has been selected for the Boeing SST airplane. This sytem is tentatively comprised of four channels, each rated 60 kVA. The first application of a VSCF system, however, will go into production on the Lockheed S-3A airplane for the Navy. This system employs two 75-kVA VSCF cycloconverter systems.

Emergency power supply.--The dc power needed for flight applications is generally obtained by rectification of the available ac power. In an emergency case (ac power supply failure), dc power is obtained from the standby batteries. Much research and development effort is presently expended on batteries. Silver-zinc and zinc-air batteries have high power densities and are hopeful possibilities for future aircraft use (refs. 15 and 16).

In emergencies, ac power is obtained from the batteries through static inverters (ref. 17.). If dc power can be generated in large quantity and with lower system weight, the ideal situation is to use dc as the main power supply and obtain whatever ac power is needed through the lightweight inverters.

Many unconventional techniques have been developed to generate dc power in the space programs. Some of these static or direct methods of electrical power generation offer good prospects for the aircraft application. Aluminum cells have a potential for an energy density of 200 W-hr/lb at the 100-hr discharge rate. Other prospects are thin-film cells, MHD, thermionic-converters and fuel cells. Fuel cells are especially promising if they can use the same fuel as the aircraft engines (refs. 18, 19, and 20).

As digital computation finds increased use in aircraft, the demand for an uninterruptable power supply becomes more pressing. Electronic gear such as the autopilot, inertial navigation, air data computer and automatic landing is extremely sensitive to voltage transients. Voltage overshoots can be effectively eliminated by surge suppressors, zener diodes, etc., whereas voltage dips present a more serious probelem. Energy stored by capacitors may fill in gaps of up to 1-msec duration. It is not economical, however, to supply stored energy for longer times (up to 10 msec). The seriousness of the effect on short duration power discontinuity is particularly evident during operation of the aircraft automatic landing system where the vehicle goes through a critical flight phase. Fortunately the amount of power required for uninterruptable service is relatively small, in the order of 1 percent of the total installed generating capacity or less. The concept of uninterruptable power has been already realized for ground computer operation. The approach there is to convert 3-phase utility power to dc. A battery is connected across the dc link which is always charged, and the dc link feeds an inverter which changes the power back to 3-phase ac. In case of failure of the ac input, the battery takes over at any part in the cycle, thereby replacing the failed source instantaneously.

One of several possible approaches for utlimate reliable ac power is schematically shown in fig. 60. An inverter is floating in parallel with the regular normal supply to the equipment. Upon loss of the regular supply source, the inverter takes over instantly. Power is prevented from flowing in the direction towards the normal supply source by semiconductor switches or magnetic decoupling. The primary supply to the inverter(s) may be constituted by all generators via individual TR units and by the airplane battery. Blocking diodes prevent power interchanges between sources.

Power Conversion and Conditioning

Solid-state and integrated circuit devices are being used in power conversion equipment for lightweight and high reliability. Higher rated and better performance items are under development. A manufacturer recently claimed to have successfully developed a high power (10 to kVA) cycloconverter with fixed-frequency sine-wave output having low harmonic contents so that large and heavy filters are not needed (ref. 21).

Solid-state EMI suppression technique is possible to reduce the weight of EMI equipment by an order of magnitude. Its development is a worthwhile effort.

Magnetic materials with high saturation flux density such as vanadium permendur have been developed in the space program. The application of these materials in the design of inductors and transformers will result in lighter weight and smaller size units. Various developments such as insulators with higher dielectric strength (H-film) and capacitors tantalum applicable in high temperature environments contribute significantly to advance the state-of-the-art of the power conditioning system (refs. 22 and 23).

Power Distribution

One important source of wire weight reduction in aircraft is signal multiplexing. Weight reduction is also possible by the adoption of smaller wire gages (down to No. 26) due to the new available higher strength copper alloy materials.

For the future generation of large transports including SST, the weight of the distribution system will be larger than in existing aircraft. A higher voltage transmission system may become necessary. Studies should also be conducted to obtain the optimum operating frequency. Ref. 18 gives an estimated optimum frequency of between 800 and 2400 Hz and an optimum voltage of 230 v to 575 v per phase for the SST (refs. 24 and 25).

Selection of the type and size of transmission wire is important for the following reasons (refs. 26 and 27):

(1) The transmission line impedance affects the performance of the electric power system, especially in transient conditions.

(2) Size of wire has considerable effect on the impedance due to the high frequency used in the line.

(3) Various conducting materials have different conductivity, specific weight and derating due to the severe environmental conditions.

(4) Losses are proportional to the impedance of the wire.

Other factors to be considered are the configuration of the transmission lines (whether four-wires with neutral return, 3-phase delta or three-wire with ground plane return or other configurations) and the protection of the system. The optimum configuration will give an efficient system and the protection provisions will ensure reliable performance. Since the distribution system extends throughout the entire aircraft an efficient system protection is essential.

The distribution system for the standby power supply should also be studied so that an optimum configuration is obtained. Loads utilizing a standby power supply should be examined to determine if that provision is truly necessary (ref. 28).

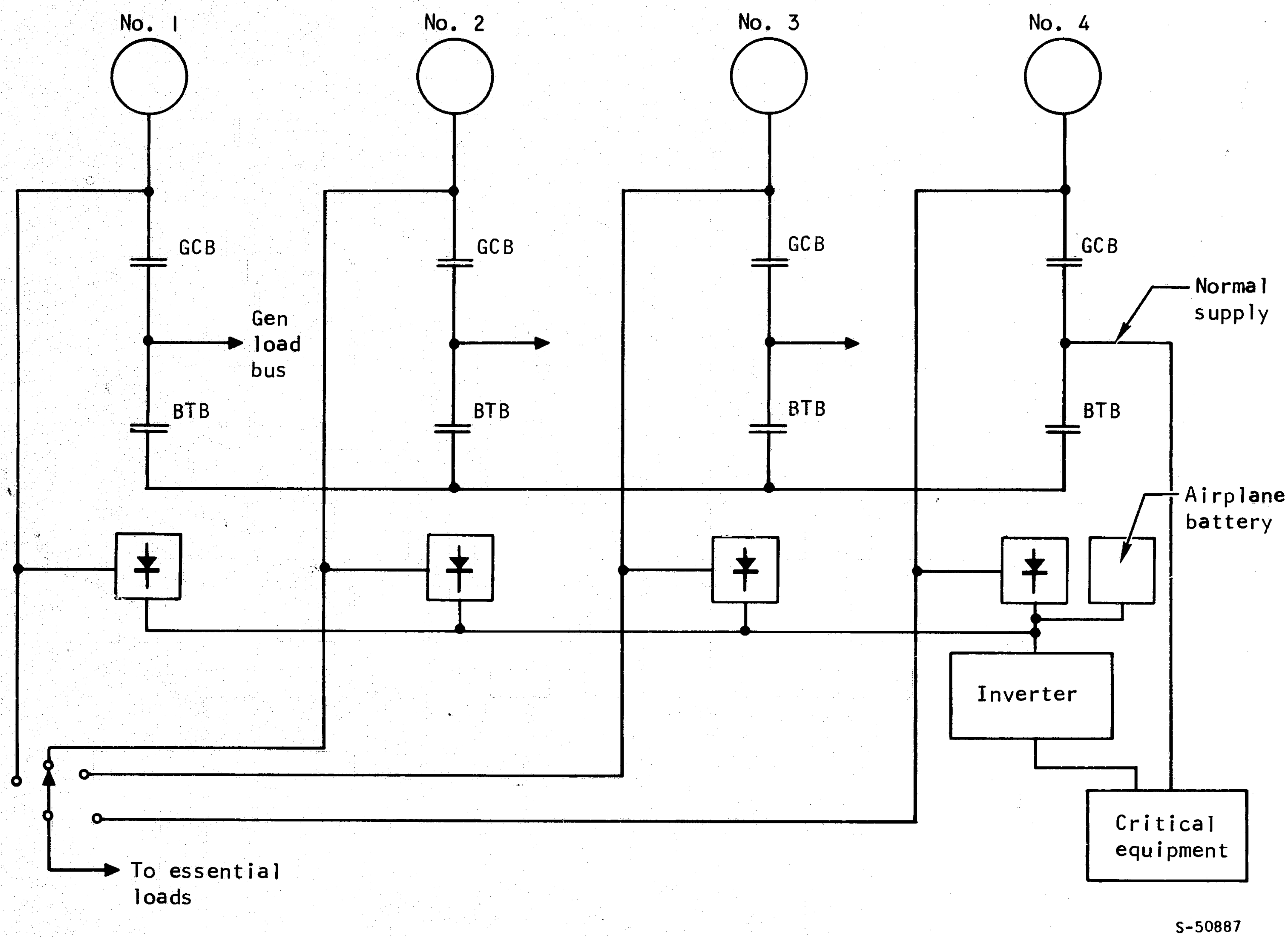

Figure 60. Ultimate Reliable Source

The problems with conventional switching devices and connectors at high altitudes are discussed in refs. 29 and 30.

Solid-state power switching and control have been under development in the last few years in LTV, Vought division. This is expected to improve the power system control and protection.

To minimize human error, fully automatic operation of the electrical system would be desirable, particularly during degraded and emergency operation of the aircraft. For example, if a three engine airplane is dispatching with one generating channel inoperative, the remaining operating generators must be capable of supplying all electrical loads continuously without using their reserve capacity (150-percent rated power for 2 min; 200-percent rated power for 5 sec). If both generators operate in parallel and for some reason a second channel becomes inoperative all connected load is suddenly dumped on the remaining generator. This generator is basically capable of supplying all loads for a minimum of 5 sec but usually will be deenergized by its undervoltage circuit within a short time. This could result in loss of the entire generating system. In critical times of serious equipment failure the flight crew will probably be occupied with a number of tasks and decisions so they may not react fast enough to deenergize all nonessential loads and relax the overloaded generator. Then the automatic load monitoring device could take on the guardian function and command corrective action. In its simplest form it could be considered as a current sensing circuit in conjunction with appropriate time delays to initiate sequencial load shedding.

Power Utilization

Much of the power utilization equipment is motors used for driving fuel pumps or other control devices. Since the motor load is an appreciable percentage of the total electrical consumption of the aircraft, the total weight of the motors is quite significant. To obtain the optimum system, a tradeoff study of the weight versus the operating frequency and voltage should be conducted, taking the load into consideration (refs. 31 and 32).

The effects of the severe environmental conditions on the motors or rotating machines in general are discussed in the literature. Most of the problems are related to the conductors, insulators, bearings, and magnetic materials (ref. 33).

Some of the loads require a stable power supply, hence a regulating system is necessary. Lightweight and reliable components for switching, sensing, and regulating have been developed using solid-state and magentic devices (ref. 34, 35, and 36).

There are numerous new sophisticated items of avionic equipment that have been developed in the last few years. The airlines are reluctant, however, to adopt these advanced airborne instruments for the following reasons:

(1) Increased maintenance cost--Although the new instruments are more reliable (reducing the frequency of maintenance), this does not offset the increase in cost because greater skill and more time are required to service these more complex items.

(2) High cost--Since avionic equipment configurations vary considerably between manufacturers, once a certain model is installed in the aircraft, the manufacturer holds a price monopoly of this equipment because it cannot be replaced by other models.

The trend is toward interchangeable avionic equipment. The airlines have pushed this concept for years to stimulate competition and reduce the price. Leading this drive is Aeronautical Radio Inc. (ARIONIC) and its associate, Airlines Electronic Engineering Committee (AEEC).

Supersonic Transport Aircraft

The anticipated environment and performance of the SST are well established (ref. 37). The major differences between the SST and existing commercial aircraft are the SST's high cruising altitude (70,000 ft vs 35,000 ft) and speed (Mach 2.2 vs Mach .6) in comparison to conventional aircraft. Thus inside the unregulated SST environment, the pressure will be lower (.7 psi vs 3.6 psi) and the temperature (-54^0 to 260^0C vs -54^0 to 50^0C) will be higher. Since much of the electrical equipment is installed in the unregulated environment, the increase in temperature can cause significant problems (refs. 38 and 39).

The safety of an aircraft is of utmost importance. One of the hazards is fire. Fire problems become particularly aggravated at high cruising speeds (Mach 7 or higher) when the environmental temperature is high enough to promote a fire. Hence in the SST electrical system design, considerations are given to (ref. 40):

(1) Prevention of fire

(2) Prevention of spread of fire

(3) Fire detection

(4) Fire extinguishment

Hydraulic System

Most of the hydraulic devices are used for loads having sudden peak demands that occur mainly in the takeoff and landing phases of the aircraft flight. In some locations both hydraulic (for short high-power bursts) and electric (for lower loadings occurring throughout the flight) motors are provided. A weight and performance tradeoff study should be conducted to determine if it is possible to eliminate one of the motors so that the load is always serviced by the same type of power. In addition, the study should consider localized hydraulic

supply centers driven by an electric motor as a possible replacement for the present hydraulic pumps which are usually mounted near the aircraft engines. Such supply centers would eliminate much of the weight penalty attributable to hydraulic line runs throughout the aircraft.

Investigations are underway for using high pressure hydraulic system in aircraft. It has been found that the main weight saving to be obtained with higher pressure hydraulic systems results from a reduced volume of hydraulic fluid in the system (Lebold and Garday "Pressure Considerations in Designing the Hydraulic System" SAE A-6 Committee Symposium, 1967).

One of the significant development efforts in hydraulic flight control system is the "fly-by-wire" system. Fly-by-wire concepts are currently being considered by McDonnell-Douglas, Sperry, and Air Force Flight Dynamics Laboratory. Space vehicles have utilized the FBW concept as have the XV-4 and X-15. The Concorde will utilize the concept.

Recent Views on Reliability

To study the reliability of aircraft electrical power systems, certain criteria for reliability should be established. Also, certain concepts understood in general by electrical engineers in the aerospace field should be defined more precisely to indicate their importance to reliability, safety, maintenance, and the cost of ownership.

It would be dangerous, for example, to adopt a technically correct but misleading procedure of reliability engineering, even though it is almost universally employed for the evaluation and comparison of redundant systems. By definition, redundant systems are not incapacitated by one or more failures. Often, however, reliability people compute an equivalent single-thread system MTBF from estimated component failure rates. This latter value is then used for comparative evaluation of competing system configurations.

The trouble results from calculations based on relevant probabilities for one flight, as illustrated by earlier SST MTBF estimates. The wingsweep angle-changing mechanism for the earlier configuration of the Boeing SST was estimated at 14,000,000-hr MTBF based on a 2500-hr MTBF value for each redundant channel. This value no longer seems high when the life of the airplane is used for computation instead of a single flight.

Simpler, more valid computation techniques are available. One of the best methods is to compare competing redundant system concepts and designs with a standard set of criteria such as those presented below.

(1) Total installed weight and associated aircraft penalty, a direct function of degree and sophistication of system redundancy and designed overcapacity.

(2) Initial cost of installed system, a direct function of degree of system redundancy.

(3) Mean aircraft flight time between component unscheduled removals (MTBUR) and resultant maintenance actions, usually a direct function of the degree of system redundancy.

(4) Dispatch reliability, a direct function of the degree of redundancy, is a probability value commonly expressed by its complementary probability of delay in number of delays or cancellations per year or per arbitrary number of scheduled dispatches or departures. A recent study conducted by Garrett/AiResearch for Airbus disclosed that cancellation of one DC-10 or L-1011 flight involved a loss of revenue to an airline of $56,000.

(5) Inflight reliability or the probability of not aborting to a field other than the scheduled destination; this parameter is closely associated with dispatch reliability as it implies system tolerance of two or more component failures, one before flight and one (or more) during flight.

(6) Safety of flight, which defines the requirements for the emergency system (dc battery, deployable ram air turbine, onboard auxiliary power unit, etc.).

(7) Accessibility or required access time to prepare to replace failed components, always additive (in time) to line maintainability.

(8) Line maintainability or the time required to replace a failed component with a spare, and thus restore a system to its original fully operational state; time for access is never included here.

(9) Shop and depot maintainability or the required time, technical manuals, trained personnel, test equipment, and spare parts required to repair failed or worn components, whether they be LRU (line replaceable units) or an entire subsystem; this parameter, expressed in units of maintenance man-hours per flight hour, is commonly translated into direct labor cost (dollars per flight hour). This criterion is sufficiently important to be specified for new aircraft systems. Airlines are now requiring their airframe suppliers to guarantee maximum values of shop and depot maintenance, and suppliers, in turn, are placing similar requirements on their component and subsystem suppliers.

(10) Failure prediction, detection, and compensation (failure diagnosis, isolation, and load reconnection), a necessary function involving all the previously enumerated functions.

Each of the above parameters is important. Some, such as accessibility, present acute problems; all present engineering problems of the most practical type with profound economic impact.

There is no accepted standard on electric power channel capacity at the present time. Recent thinking on minimum allowable channel capacity is outlined in Table 26.

TABLE 26

RECOMMENDED AND MINIMUM ALLOWABLE CHANNEL AND SYSTEM DESIGN CAPACITIES

Number of engines	Number of channels	Design channel capacity	Dispatch with one channel inoperative	Abort with one channel inoperative	Comments
4	4	.25	No	No	No sufficient margin for commercial aircraft
		.333	Yes	No	Minimum contingency factor
		.50	Yes	No	Adequate margin for most emergencies
3	3	.333	No	No	Second failure critical (see text)
		.50	No	No	Second failure critical (see text)
		.75	Yes	No	Unacceptable contingency factor
		.67 N/1.0E	Yes	No	Adequate margin for most emergencies (Note assumption that 1/2-hr emergency capacity = 1.5 normal)
2	2	.50	No	No	Second failure critical
		.67	No	No	Unacceptable contingency factor
		.57 N/1.0E	Yes (if APU is running and has the same capacity as the main generator)	No	Acceptable with 1/2-hr emergency rating of 1.5 (.67) Adequate margin without flight engineer

Notes: N = Nominal continuous capacity
E = Suggested emergency 1/2-hr rating = 1.5 N

Concurrent with this study the feasible reliability goals for each item of hardware associated with candidate channel, subsystem, or system design should be established. This will allow quantitative values to be clarified for those dependent parameters listed such as dispatch and inflight reliability, line and shop maintainability, and MTBURs.

The mathematical relationships required for this work are too well known to repeat. The method of establishing values for the reliability performance of new concepts and conceptual hardware should be discussed, however, since little emphasis is placed on the necessary mathematical connection between MTBF, MTBUR, time between scheduled overhaul (TBO), and design life in the available literature.

MTBF is a quotient of cumulative flight or equipment operating time divided by cumulative confirmed failures. MTBUR is simply the quotient of cumulative flight or equipment operating time divided by cumulative removals. This frequently significant difference between MTBF and MTBUR reflects deficiencies in effectiveness of failure detection, isolation, and identification.

Time between scheduled overhaul (TBO) is an arbitrary but essential definition of scheduled remanufacture to nearly a "zero time condition."

Design life is a term rarely defined and frequently confused with service life. The latter term impliedly includes scheduled and unscheduled overhaul (remanufacture) whereas design life is usually employed synonomously with time before overhaul.

Using the above terms in conjunction with the concept that "for any population of hardware, some fail prematurely or before the design life (TBO)", the probability of failure occurring is $\left(1 - e^{-\frac{TBO}{MTBF}}\right)$. For very rugged components (a heavy-duty gearbox, for example) the MTBF will tend to exceed the TBO several times. For highly stressed or very complicated hardware (some constant speed drives, for example) the TBO tends to exceed the MTBF; the implications are obvious.

The following inferences result when the ten criteria listed above are applied to the electrical systems:

(1) The overall penalty resulting from total installed weight and associated aircraft penalties is an inverse function of aircraft optimum cruise lift over drag ratio. This ratio, which approximates 16 for current high performance subsonic transports, will be reduced to approximately 8 for SST; thus, installed weight is of sharply increased significance for SST.

(2) The total initial cost of newly conceived power generation and distribution systems is appreciably higher than the cost of existing state-of-the-art components. To compensate for this initial cost, a reduction in cost of ownership will be necessary, highly desirable, or both. Such reduction is most likely to result from increased reliability, plus reduced incidence of failure and reduced line and shop maintenance costs. Compensation for initial cost in this manner appears feasible for VSCF and dc power.

(3) The VSCF approach is potentially more reliable than the SCD approach. The VSCF approach has a weight disadvantage that is not pronounced and can be improved.

(4) The dispatch reliability of different systems can be made to correspond to any desired value, however, difference levels of system redundancy (weight and costs) may be required.

(5) Comment (1) also applies to in-flight system reliability.

(6) Safety of flight may depend upon a good dc emergency source. The improved dc power generation technique employed in past and present manned spacecraft appears to be the best emergency source in addition to possessing the greatest growth potential for primary power generation.

(7) Accessibility is such a difficult problem that any improvement in the SST in this area is advantageous. Avoidance of a CSD falls within this concept, not only for accessibility but also for most of the ten previously listed criteria.

(8 and 9) No comment is required on these two criteria, except that both would benefit from improved reliability.

(10) Since fault (failure) detection, isolation, and the identification concept are part of the loop, complexity can pose a serious reliability problem. In aircraft air conditioning systems, no attempt is made to fault isolate certain components because the failure frequency of the fault detection and isolation equipment would be appreciably greater than that of the monitored component (e.g., a heat exchanger). Sophisticated mathematical analysis and optimization may lend itself to solving the problem associated with this task and may conceivably become an important element in the final choice of a prime candidate mode of generation and distribution. Current work on fault detection and isolation on major subsystems of the McDonnell Douglas DC-10 large commercial jet transport clearly indicates a void in the scientific approach to optimization of fault detection and isolation. The methodology and techniques now employed can only be described as primitive and routine. Some of the built-in test equipment (BITE) has more than doubled the failure rate of certain electronic components, and extensive redesign has been required to reduce the complexity.

Recommended Research and Development Efforts

As a result of the present study activities, the areas which appear likely to receive further research and development attention are listed below:

(1) Automatic load management

(2) Connectors

(3) Uninterruptible power system for airborne digital controls

(4) Signal multiplexing techniques

(5) Solid-state EMI suppression

(6) Brushless dc motor

(7) Dc circuit breakers and contactors

(8) Adaptation of dc transformer in aircraft

APPENDIX A

POWER CHARACTERISTIC AND SYSTEM COMPONENT REQUIREMENTS

To illustrate the design requirements for a typical aircraft electrical power system, the following information is abstracted from a design specification for the Lockheed C-141 aircraft.

General System Requirements

The primary ac power system is a 3-phase, 4-wire "Y" system with a nominal voltage of 115/200 V and a nominal frequency of 400 Hz. The neutral point of the power source is connected to ground, and the ground is considered the fourth conductor. The primary ac power source is four constant frequency generating systems or channels, each one associated with each of the four aircraft engines. These four channels may have any one of three operation modes: (1) normally operated in parallel; (2) not in parallel but nominally synchronized; or (3) not in parallel and not synchronized. The line drop to equipment supplied from this system is less than 4 V for continuous duty equipment and 8 V for intermittent duty equipment.

The dc power system is a 2-wire, grounded system with a nominal voltage of 28 V. The negative of the power source is connected to ground, and the ground is considered the second wire. Unregulated transformer-rectifiers (TR's) supplied from the primary ac power system provide the dc power source. The voltage variation during operation of the dc utilization equipment includes ac line voltage drop, TR voltage regulation, and dc line voltage drop. Utilization equipment is maintained at specified performance levels; the power system characteristics are listed below under Detailed System Requirements.

When it is necessary to use power with characteristics or tolerances other than those provided, conversion to those characteristics or tolerances is accomplished by the utilization equipment.

Detailed System Requirements

Ac power system characteristics.-- The steady-state phase voltage for single and three phase limits are given in table A-1. These limits are applicable during 380- to 420-Hz operation.

The displacement between adjacent phases is within the 120 ±1.5-deg limit. Maximum spread in phase voltages is less than 3 v for all aircraft operations. The voltage waveform is within the following limits:

(1) Crest factor--1.41 ±0.1

(2) Total harmonic content--4 percent of the fundamental (rms), when measured with a distortion meter as distortion of the fundamental frequency.

TABLE A-1

STEADY-STATE AC VOLTAGE LIMITS

	Single-phase limits	Average of three-phase limits
	115-V equipment	115/200-V equipment
Continuous duty equipment	107.5 to 119.5	108.5 to 117.5
Intermittent duty equipment	103.5 to 119.5	104.5 to 117.5

(3) Individual harmonic content--3 percent of the fundamental (rms), when measured with a harmonic analyzer.

The voltage modulation does not exceed a 3.5-V amplitude when measured as the peak-to-valley difference between the minimum voltage and the maximum voltage reached on the modulation envelope over at least a 1-sec period. The voltage modulation is illustrated in fig. A-1. The frequency components of the voltage modulation envelope waveform are within the limits specified in fig. A-1.

Voltage transients are within the limits of fig. A-2 for all operations of the aircraft electric system. The most severe phase transient is used in determining conformance to fig. A-2. The voltage transients for all normal electric-system operations are within limits 2 and 3 of fig. A-2. The ac voltage transients which result from abnormal electric-system operation are less than limits 1 and 4 of fig. A-2.

The ac power systems frequency is maintained at 400 ±4 Hz for steady-state operation. Variation within steady-state frequency limits owing to drift is not more than ±2 Hz for any period of steady-state electric-system operation. Frequency variation owing to drift does not occur at a rate greater than 15 Hz/min.

Variations of frequency owing to frequency modulation during any 1-min period are within a band of ±2 Hz about a mean frequency. The mean frequency may drift within the limits defined in the above paragraph. Rates of frequency change owing to frequency modulation do not exceed 13 Hz/sec.

Frequency transients are confined within the limits of fig. A-3 for all aircraft operations. Rates of frequency change during a transient will not exceed 500 Hz/sec for any period longer than 15 msec. Frequency transients resulting from normal system operations are within limits 2 and 3 of fig. A-3. Those resulting from abnormal electric-system operations are within limits 1 and 4 of fig. A-3.

The electric distribution and utilization systems will have a phase sequence A, B, C corresponding to T_1, T_2, T_3 of the power source.

Primary ac power system

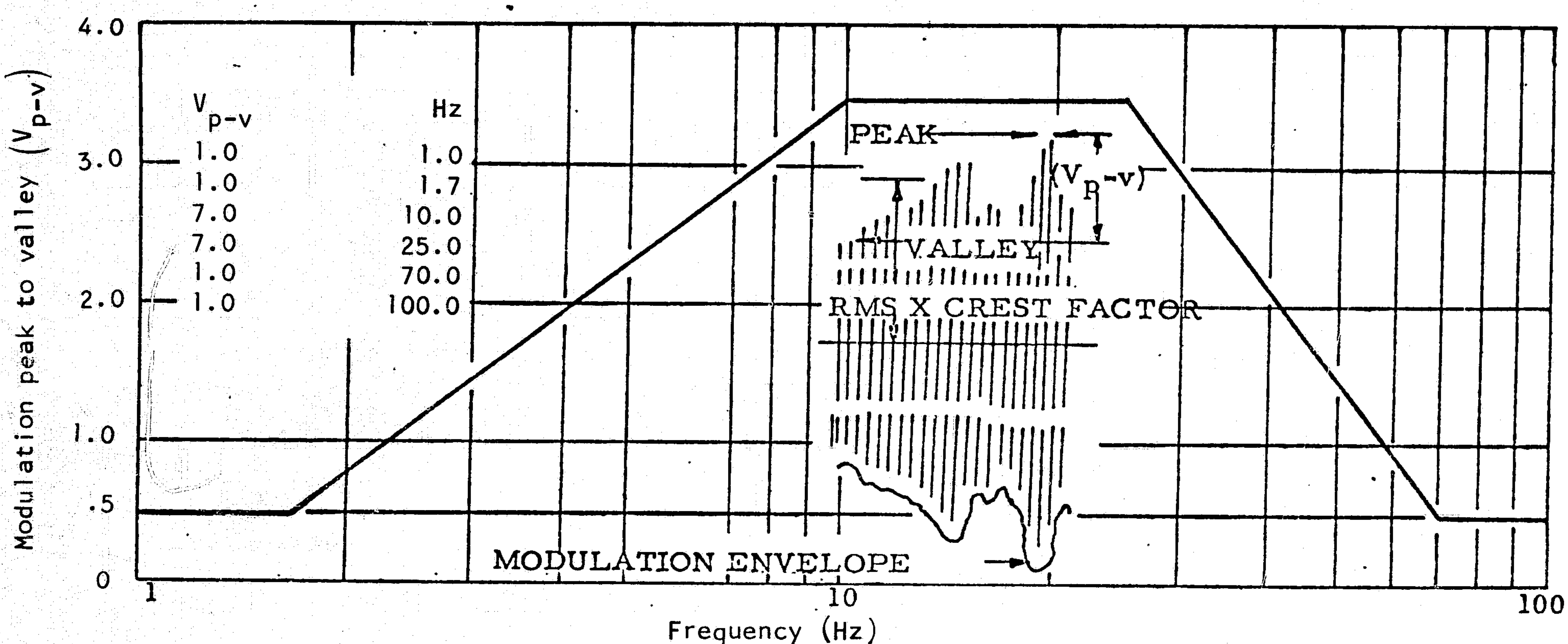

Figure A-1. Frequency Characteristics of ac Voltage Modulation Envelope

Primary ac power system

LIMIT 1 SEC	LIMIT 1 V_1	LIMIT 2 SEC	LIMIT 2 V_1	LIMIT 3 SEC	LIMIT 3 V_1	LIMIT 4 SEC	LIMIT 4 V_1
5.0	137	5.0	126	5.0	104	5.0	103
3.0	139	2.0	128	2.0	103	5.0	90
1.0	155	1.0	130	1.0	100	3.0	90
0.4	174	0.30	142	0.30	87	3.0	0
0.2	184	0.10	164	0.10	62		
0.1	190	0.03	173	0.05	56		
0.0	190	0.0	173	0.02	54		
				0.02	0		

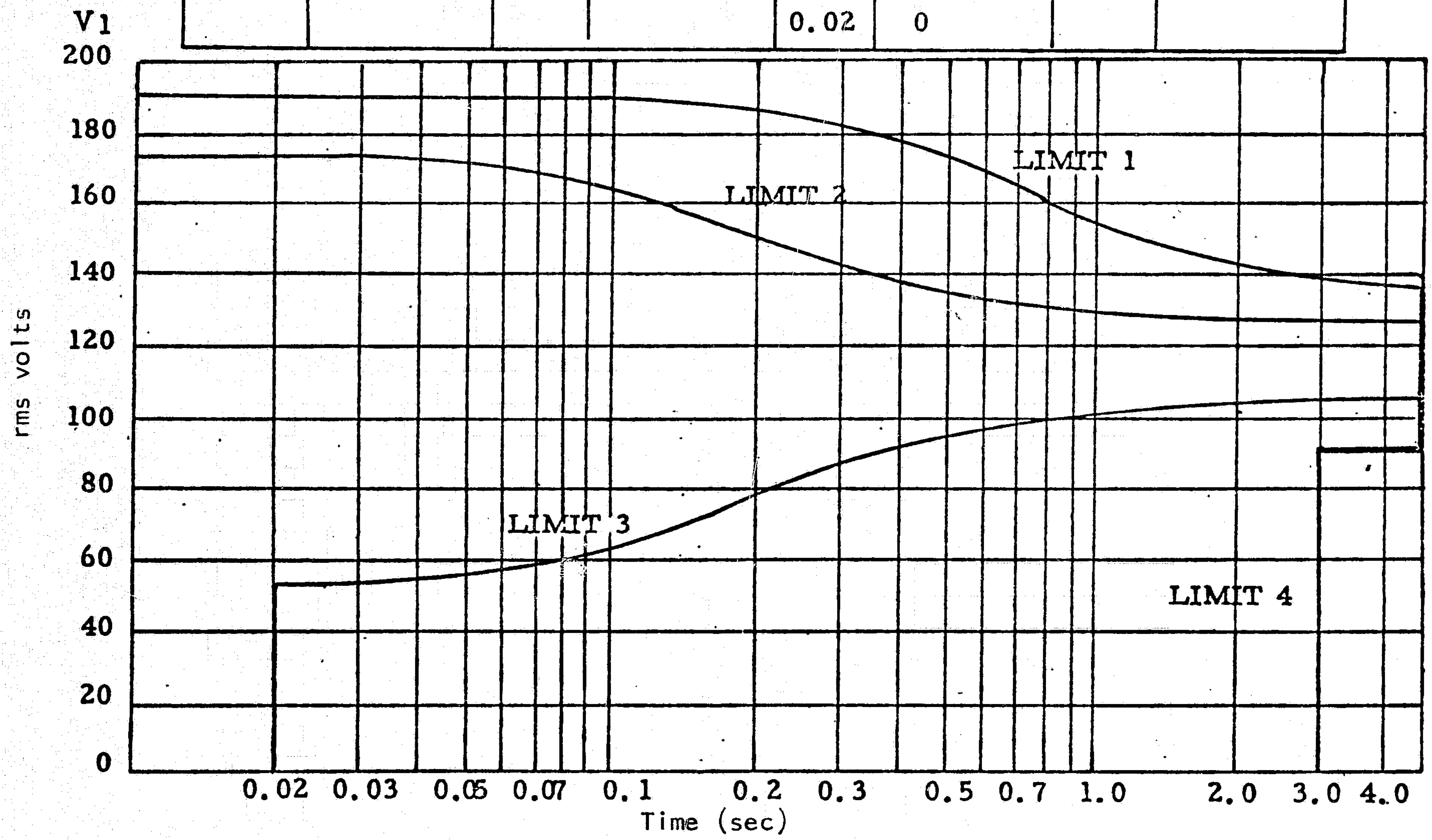

Figure A-2. Transient ac Voltage Limits

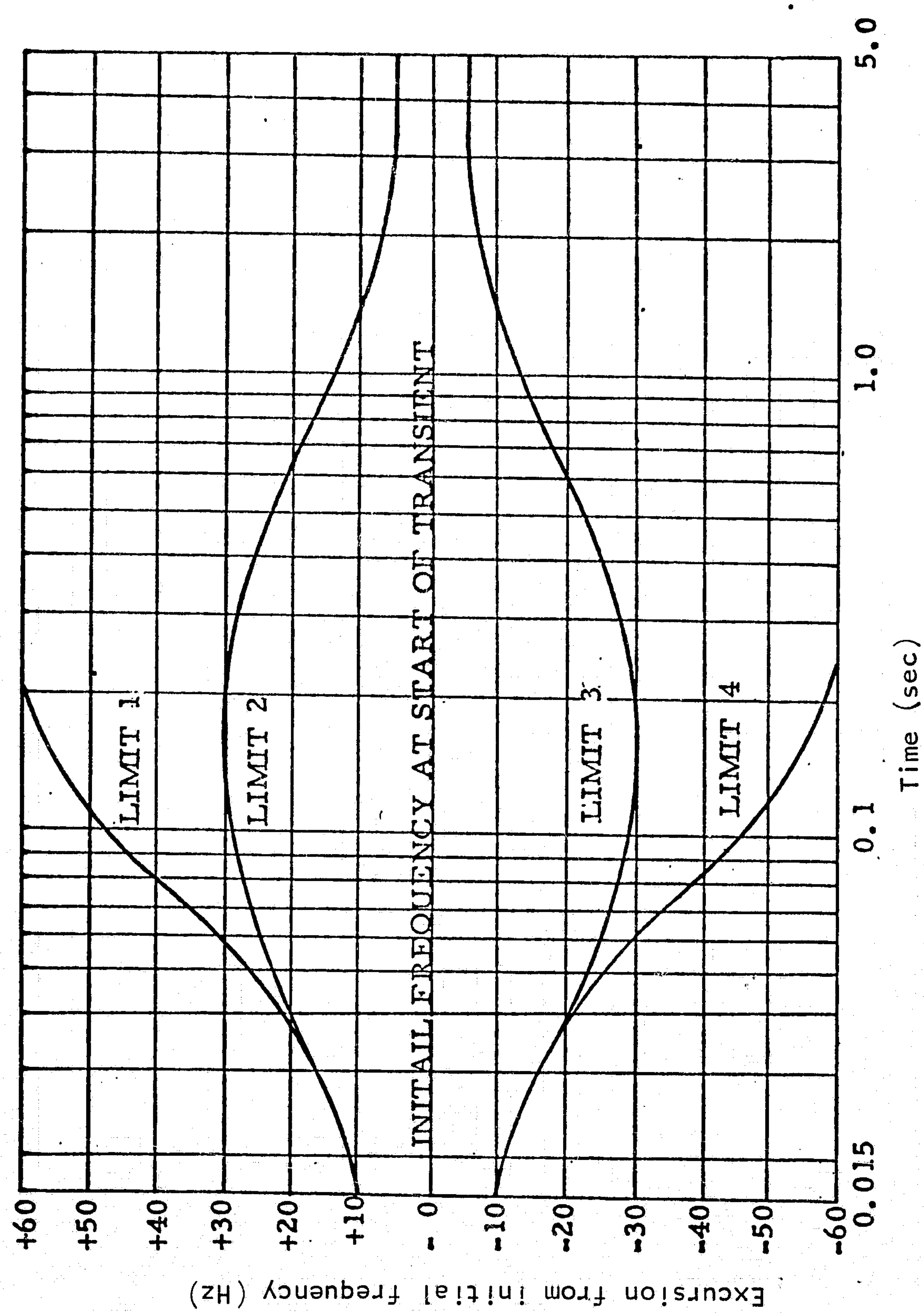

Figure A-3. Transient Frequency Limits

LIMIT 1		LIMIT 2		LIMIT 3		LIMIT 4	
SEC	FREQ	SEC	FREQ	SEC	FREQ	SEC	FREQ
0.000	0	0.000	0	0.000	0	0.000	0
0.015	+10	0.015	+10	0.015	-10	0.015	-10
0.080	+40	0.080	+28	0.080	-28	0.080	-40
0.250	+60	0.200	+30	0.200	-30	0.250	-60
1.00	+60	1.00	+13	1.00	-13	1.00	-60
5.00	+60	3.00	+5	3.00	-5	5.00	-60

Dc power system characteristics.-- Steady-state voltage shall be within the limits specified in table A-2.

TABLE A-2

STEADY-STATE DC LIMITS

Continuous duty equipment	24 to 28.5 v
Intermittent duty equipment	23 to 28.5 v

The ac peak of ripple voltage to average dc voltage is less than 1.5 V when measured with a peak reading vacuum tube voltmeter in series with a 4.0-μF capacitor. The higher of the two values measured when the voltmeter is successively connected for each of two polarities is considered the ripple voltage. The frequency components of the ripple are within the limits given in fig. A-4 when measured as conducted interference.

Voltage transients are within the limits of fig. A-5 for all operations of the aircraft electric system. The dc voltage transients for all normal electric-system operations are limits 2 and 3 of fig. A-5; dc voltage transients for abnormal electric-system operation are less than limits 1 and 4 of fig. A-5.

Utilization of Aircraft Electric Power

Conversion.-- Equipment which requires conversion of input power to power with other characteristics accepts the power as defined for modification and use. Modification and use is integral with the utilization systems or utilization equipment. Utilization equipment of a size to require ac input power above 500 VA may include integral static converters to obtain up to 5 A of 28-Vdc power in lieu of requiring the 28-Vdc power as specified.

Normal electric-system operation.-- During normal operation of the electric system, utilization equipment:

(1) Provides 100-percent performance, except when the detail specification for a given utilization equipment defines specific regions of the electric system characteristics with corresponding degrees of performance degradation.

(2) Remains safe.

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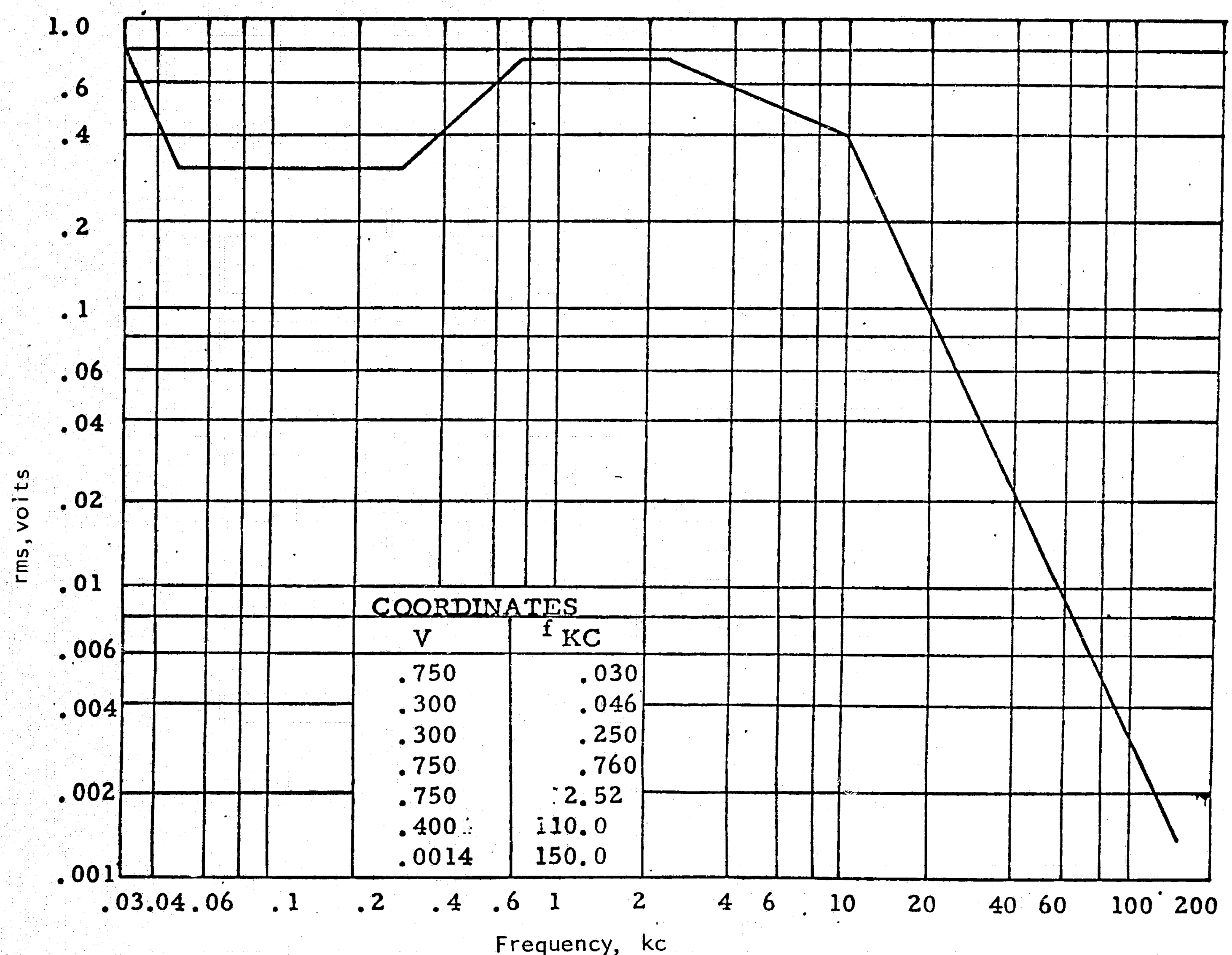


Figure A-4. Frequency Characteristics of Ripple in 20 Vdc Electric Systems

LIMIT 1		LIMIT 2		LIMIT 3		LIMIT 4	
SEC	VOLTS	SEC	VOLTS	SEC	V_1	SEC	V_1
5.0	34	5.0	31.5	5.0	21.7	5.0	21.5
3.0	35.0	2.0	32.0	2.0	21.5	5.0	18.8
1.0	39.0	1.0	32.5	1.0	20.9	3.0	18.8
0.4	43.5	0.30	35.5	0.30	18.2	3.0	0
0.2	46.0	0.10	41.0	0.10	12.9		
0.1	47.5	0.03	43.3	0.05	11.7		
0.0	47.5	0.0	43.3	0.02	11.3		
				0.02	0		

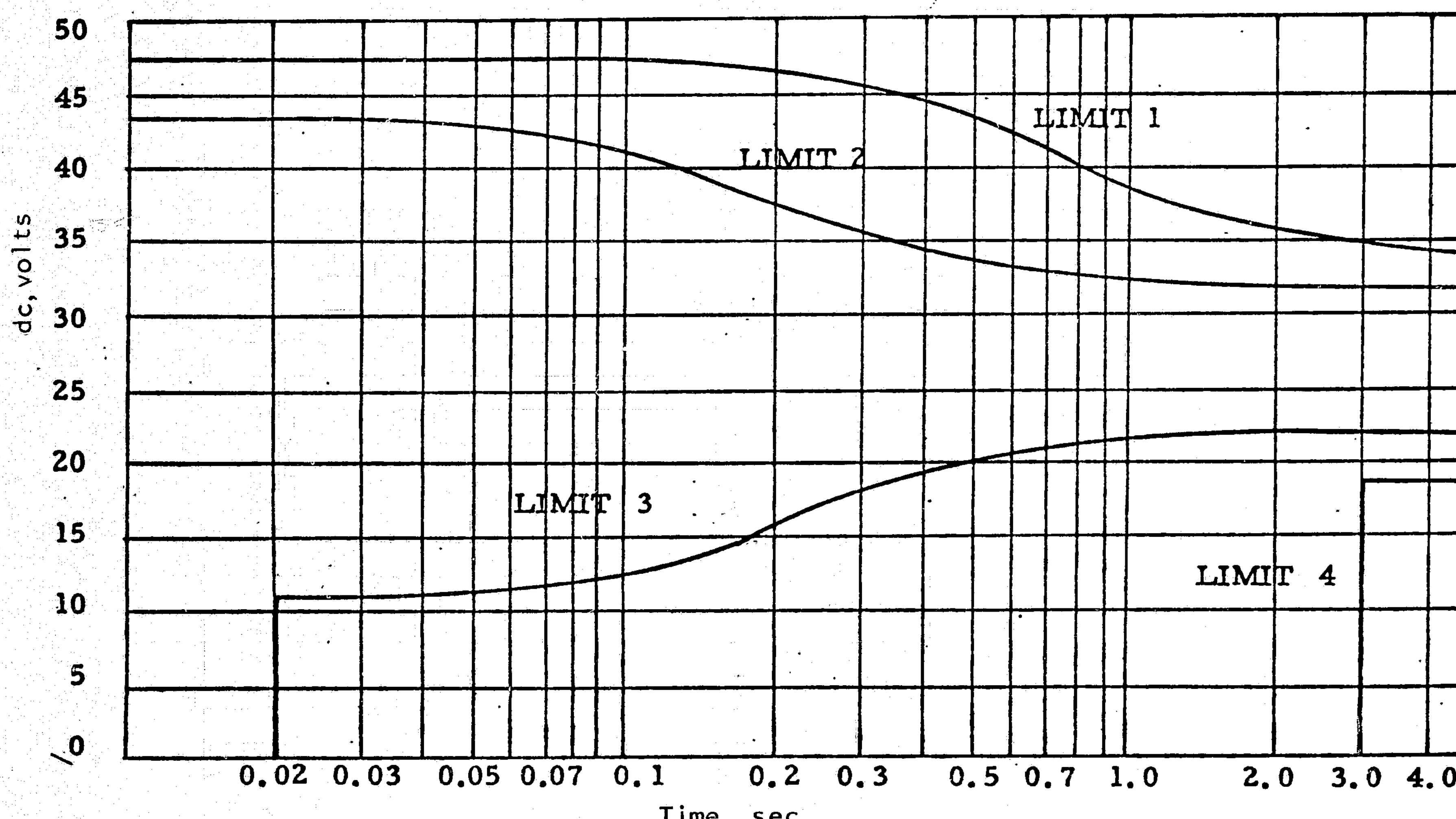


Figure A-5. Transient dc-Voltage Limits

(3) When degraded performance has been permitted for given regions of given characteristics, after operation in such regions with return to other regions of normal electric-system operation, the utilization equipment:

(a) Automatically recovers to 100-percent performance.

(b) Remains unaffected in reliability.

Abnormal electric-system operation.-- During abnormal operation of the electric-system, utilization equipment:

(1) Has no performance requirements, unless the detail specification for a given utilization equipment requires specific degrees of performance to be maintained within specific regions of the electric-system characteristics.

(2) Remains safe.

(3) May have momentary loss of function; however, this momentary loss does not affect later equipment performance.

(4) After abnormal operation of the electric-system and with return of the electric-system to normal operation, utilization equipment:

(a) Recovers automatically to specified performance, unless the detail specification for a given utilization equipment permits manual reset of equipment after the abnormal electric-system operation.

(b) Has negligible effect on reliability owing to the abnormal electric-system operation.

Voltage transients.-- For testing performance of utilization equipment during conditions of input voltage transients, voltage transients are considered as any voltage as its corresponding time on the limits of fig. A-2 and A-5.

Warmup.-- Time required for equipment to warmup prior to obtaining specified performance is minimized. Time to return to specified performance, after a power interruption, does not exceed the actual thermal or mechanical requirements, or both. Warmup time is less than 5 min.

Influence on electric system.-- There is no influence by utilization equipment on the power characteristics at the terminal inputs which would cause these characteristics to go beyond the limits specified.

The modulation induced by varying loads within utilization equipment does not cause voltage modulation or ripple to go beyond the limits at the terminals of the utilization equipment given above. This self-modulation is caused by variations in the current (required by the equipment) that cause a varying voltage drop in the wiring of the power circuit to the equipment.

Ac power.-- For loads less than 500 VA, three-phase power is used when practicable; for ac input demands exceeding 500 VA, three-phase power is required. The averages of three-phase steady-state voltage limits in table A-I are applicable only when a phase of the three phases is not utilized as a single-phase load. For ac input demands not exceeding 500 VA, the equipment may require single-phase power. Equipment which is inherently single phase in power consumption presents, if practicable, a three-phase demand by being internally segregated into three single-phase loads. Single-phase power is used only on a line-to-neutral basis.

Equipment requiring three-phase power requires equal phase volt-amperes and power factor insofar as practicable. The phase volt-ampere difference between the highest and lowest phase values, assuming balanced voltages, is less than the limits specified in fig. A-6.

Equipment utilizing ac power is designed to present as near a unity power factor as practicable for all modes of equipment operation. The fully loaded equipment presents a power factor on the worst phase not less than the limits specified in fig. A-7.

One phase of three-phase power can fail without resulting in an unsafe condition. During failure of the one phase, no equipment performance is required unless specified in the equipment detail specification.

Power failure.-- For equipment requiring both ac and dc power, one power source may fail without resulting in an unsafe condition. During this loss of power no equipment performance is required. Where performance is not required but power is required to maintain equipment standby readiness, the standby power requirement is minimized.

Power tolerance.-- Input power requirements do not vary by more than +10 percent and -10 percent of an established limit between production units of a given utilization equipment.

Definitions

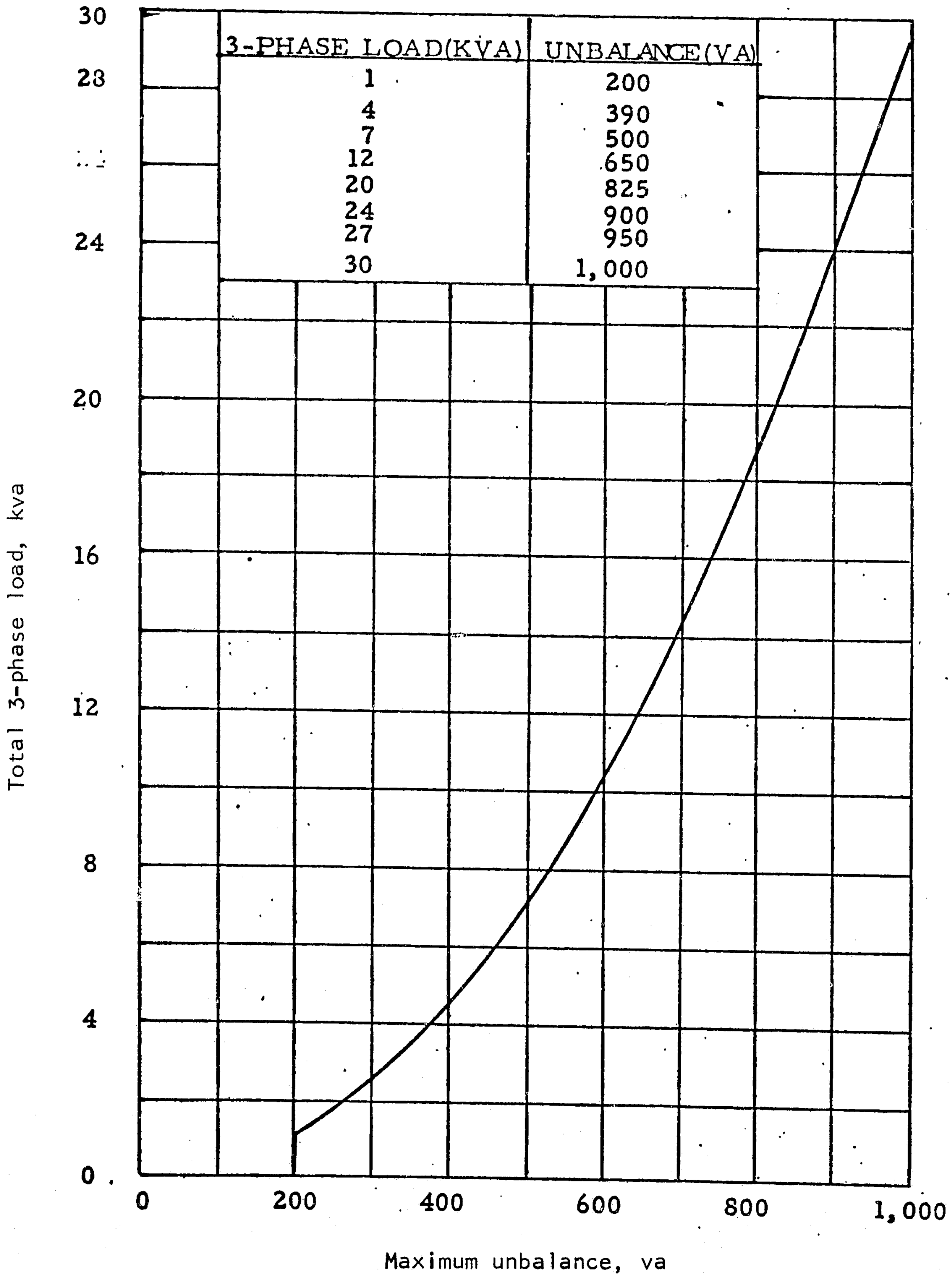
A listing of pertinent definitions is presented below.

(1) Average value--The arithmetical sum of the phase values divided by the number of phases.

(2) Ground--The referenced ground for the negative of the dc and the neutral of the ac in the power generation and power utilization systems.

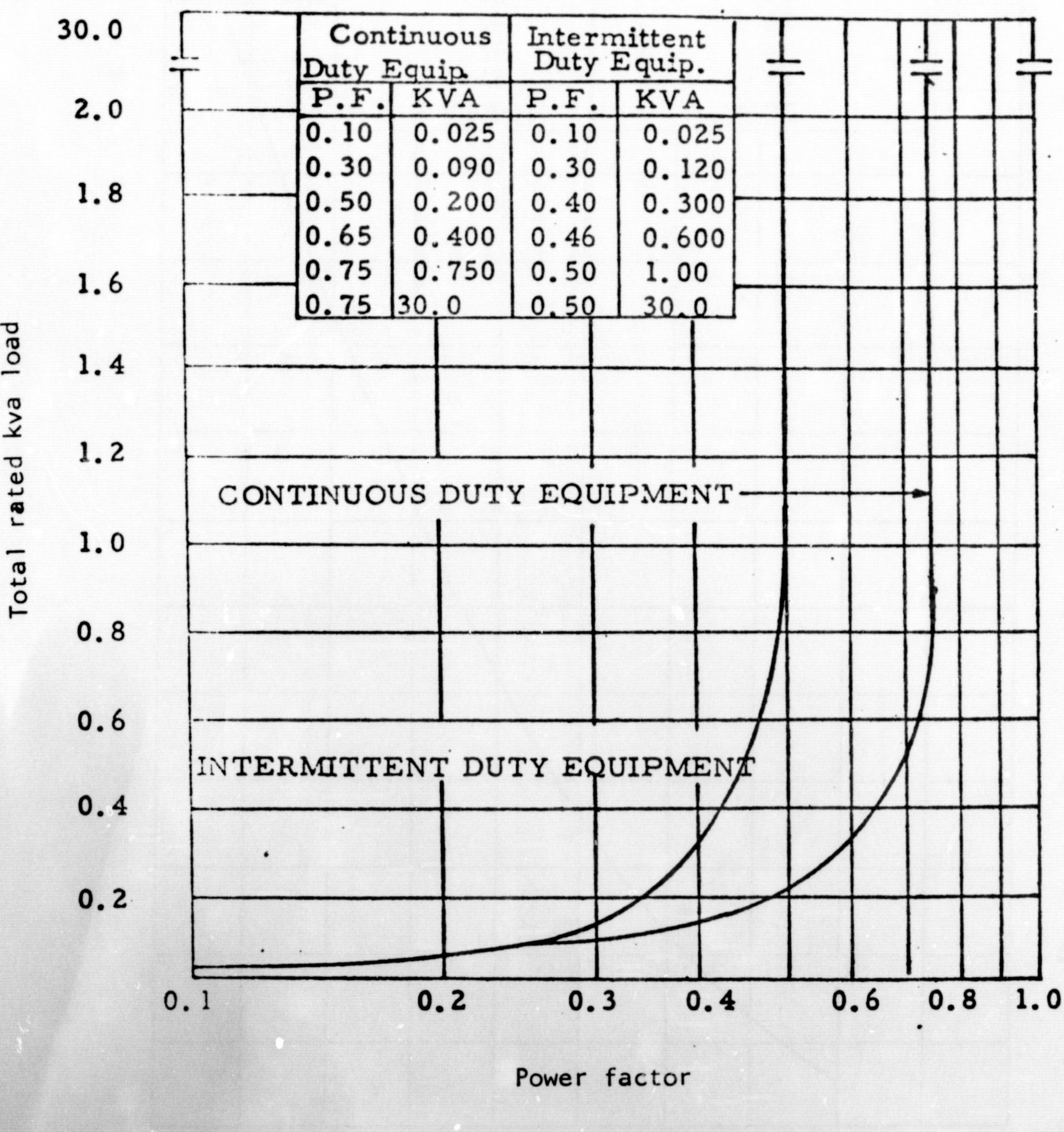
(3) Transient--The changing condition of a characteristic which goes beyond the steady-state limits and returns to the steady-state limits within a specified time period.

(4) Total harmonic content--The total rms voltage of a complex wave remaining when the fundamental component is removed.



3-PHASE LOAD(KVA)	UNBALANCE(VA)
1	200
4	390
7	500
12	650
20	825
24	900
27	950
30	1,000

Figure A-6. Unbalance Limits for 3-phase Utilization Equipment



Continuous Duty Equip.		Intermittent Duty Equip.	
P.F.	KVA	P.F.	KVA
0.10	0.025	0.10	0.025
0.30	0.090	0.30	0.120
0.50	0.200	0.40	0.300
0.65	0.400	0.46	0.600
0.75	0.750	0.50	1.00
0.75	30.0	0.50	30.0

Figure A-7. Lagging Power Factor Limits for Utilization Equipment

(5) Frequency modulation--The cyclic or random dynamic variation, or both, of instantaneous frequency about a mean frequency during steady-state electric-system operation. The frequency modulation is normally within narrow frequency limits and occurs as a result of speed variations in a generator rotor owing to the dynamic operation of the rotor coupling and drive speed regulation.

(6) Frequency modulation rate--The rate of change of frequency owing to frequency modulation when plotted against time.

(7) Frequency drift--The extremely slow and random variation in frequency within the steady-state limits occurring as a result of environmental effects and wear on the electric power-drive system.

(8) Frequency drift rate--The rate of change of frequency owing to frequency drift when plotted against time.

(9) Ac voltage values--Root mean square (rms) values for any phase of utilization equipment, a phase being considered a line to neutral.

(10) Voltage modulation--Voltage modulation is the cyclic variation or random dynamic variations, or both, about an average of the ac peak voltage during steady-state electric-system operation such as caused by voltage regulation and speed variations. The modulation envelope is formed by a continuous curve connecting each sine wave peak to the successive sine wave peak.

(11) Voltage modulation frequency characteristics--The component frequencies which make up the modulation envelope waveform.

(12) Ripple--The ac variation of voltage about a fixed dc voltage during steady-state dc electric-system operation.

(13) Unsafe condition--Any condition within the aircraft that jeopardizes the safety of the aircraft and/or the personnel aboard.

(14) Aircraft operational period--The time interval between the start of preparation for flight and the post flight engine shutdown with consequent deactivation of the aircraft electric system.

(15) Utilization equipment--Either an individual unit, set, or a complete system to which the electrical power is applied or disconnected, or both, as a whole.

(16) Intermittent duty utilization equipment--That equipment operated for 30 sec or less in any 15-min period. When the detail equipment specification does not designate otherwise, the equipment will be considered continuous duty.

(17) Line drop--The voltage difference between the point of voltage regulation and the power input terminals of the equipment.

(18) Normal electric-system operation--All functional electric-system operations required for aircraft operation, aircraft mission and electric-system controlled continuity. These operations occur any number of times at any given instant during flight preparation, takeoff, airborne conditions, landing, and anchoring. Examples of such operations are switching of utilization equipment loads, engine speed changes, bus switching and synchronization, and paralleling of electric power sources.

(19) Abnormal electric-system operation--The unexpected but momentary loss of control of the electric system. The initiating action of the abnormal operation is uncontrolled and the exact moment of its occurrence is not anticipated. However, recovery from this operation is a controlled action. This operation occurs perhaps once during a flight or it may never occur during the life of an aircraft. An example of an abnormal operation is the faulting of electric power to the structure of an aircraft and its subsequent clearing by fault protective devices.

(20) Emergency electric-system operation--That condition of the electric system during flight when the primary electric system becomes unable to supply sufficient or proper electric power, thus requiring the use of a limited independent alternate source of power.

One generator and constant speed drive is mounted on each main engine. Each of these four generators supplies its load bus through a generator circuit breaker (contactor). Each load bus is connected to a bus tie breaker (contactor) and to a synchronizing bus. A fifth generator, mounted on the auxiliary power unit (APU), is not paralleled with the others. The APU generator supplies power to the tie bus through a generator contactor only when no other power supply is connected to the tie bus.

Constant Speed Drive Requirements

Characteristics.-- The power output rating of the drive is 64 hp. The drive has a capability, however, of carrying 80 hp continuously, 96 hp for 5 min, and 128 hp for 5 sec over an input speed range of 4100 to 8500 rpm. (Cruise speed is referred to here as 7300 rpm.) An output speed of 6,000 rpm corresponds to a 400-Hz generator output.

The drive and load controller will be in accordance with the control drawing specified. Maximum is 12 x 12 x 12 in.; the controller is 3.5 x 4.5 x 6.5 in.

Component Design Requirements

The following is a summary of design requirements for the components of a typical aircraft electrical generating system, rated 40 kVA per channel. The system is comprised of:

(1) Constant Speed Drive Unit (CSD)

(2) Generator

(3) Generator Control Panel

(4) Bus Protection Panel

Applicable Government specifications are:

MIL-E-5272(ASG) (1) General Specification for Environmental Testing, Aeronautical and Associated Equipment

MIL-G-6099A (1) General Specification Generator and Regulators, Air Cooled A-C, Aircraft

MIL-STD-704 Characteristics and Utilization of Aircraft Electric Power MIL-T-7210 (USAF) (1) Current Transformer

MIL-C-26500 (USAF) (3) Connectors eneral Purpose, Miniature, Environment Resisting 200°C Ambient Temperature

MIL-L-78080 Lubricating oil, Gas Turbine Aircraft

MIL-C-8188C Corrosion Preventive Oil Gas Turbine Aircraft

MIL-I-61810 Interference Control Requirements, Aircraft Equipment

The total generating plant is comprised of four generating channels, driven by the main aircraft engines and one generating unit driven by an APU.

The maximum weight of the drive with integral oil tank is 76 lb. The maximum weight for the load controller for each drive is 5 lb.

Design and construction.-- The CSD is suitable for transmitting power from an aircraft turbofan engine pad operating at a variable speed to an aircooled generator at constant speed. The unit includes:

(1) An input Quick Attach-Detach (QAD) mounting

(2) A solenoid operated input disconnect device

(3) Malfunction detection provisions

(4) An integral speed governing system

(5) Integral overspeed protection

(6) Underspeed indication

(7) A means of permitting the generator to overrun the drive

(8) An integral oil tank

(9) A wiring harness with a single external connector

When the load is constant: over the rated input speed range, the average generator output frequency remains within the limits of fig. A-1 with changes in input speed no greater than 1 rev/sec/sec. The instantaneous output frequency during any 15sec period does not deviate more than 1 Hz from the average output frequency during that period. Frequency modulation does not exceed .1 perecent, or be at a frequency below 25 Hz.

Within the rated input speed range the sudden application or removal of any load up to 200 percent will not produce frequency variations outside the limits of fig. A-8. The frequency will return to and remain within the steady-state limits within 1.0 sec after application or removal of rated load.

Within the rated input speed range, with constant loads up to 100 percent rated, input speed accelerations up to 15 rev/sec/sec will not cause the output frequency to deviate from rated frequency by more than 2.5 percent. Combined load and speed changes will not cause greater frequency deviations than the sum of the deviations for the changes separately.

The maximum steady-state difference between the real load on any generator and the average generator real load will not exceed 6-1/4 percent of 1 generator rating for any load from zero to rated system load.

The maximum transient difference between the real load on any generator and the average generator real load with sudden application or removal of any load up to rated shall not remain above 20 percent of one generator rating for a period not to exceed 50 milliseconds, and shall not exceed 50 percent of 1 generator rating during this period.

General Component Requirement

A typical aircraft electrical generating system, rated 40 kVA, comprises a constant speed drive (CSD), a generator, a generator control panel, and a bus protection panel. Applicable design requirements for these components are provided in the government specifications listed below

MIL-E-5272C (ASG) (1) General specification for environmental testing, aeronautical and associated equipment.

MIL-G-6099Z (1) General specification generator and regulators, air cooled ac, aircraft.

MIL-STD-704 Characteristics and utilization of aircraft electric power.

MIL-T-7210 (USAF) (1) Current transformer.

MIL-C-26500 (USAF) (3) Connectors general purpose, miniature, environment resisting 200°C ambient temperature.

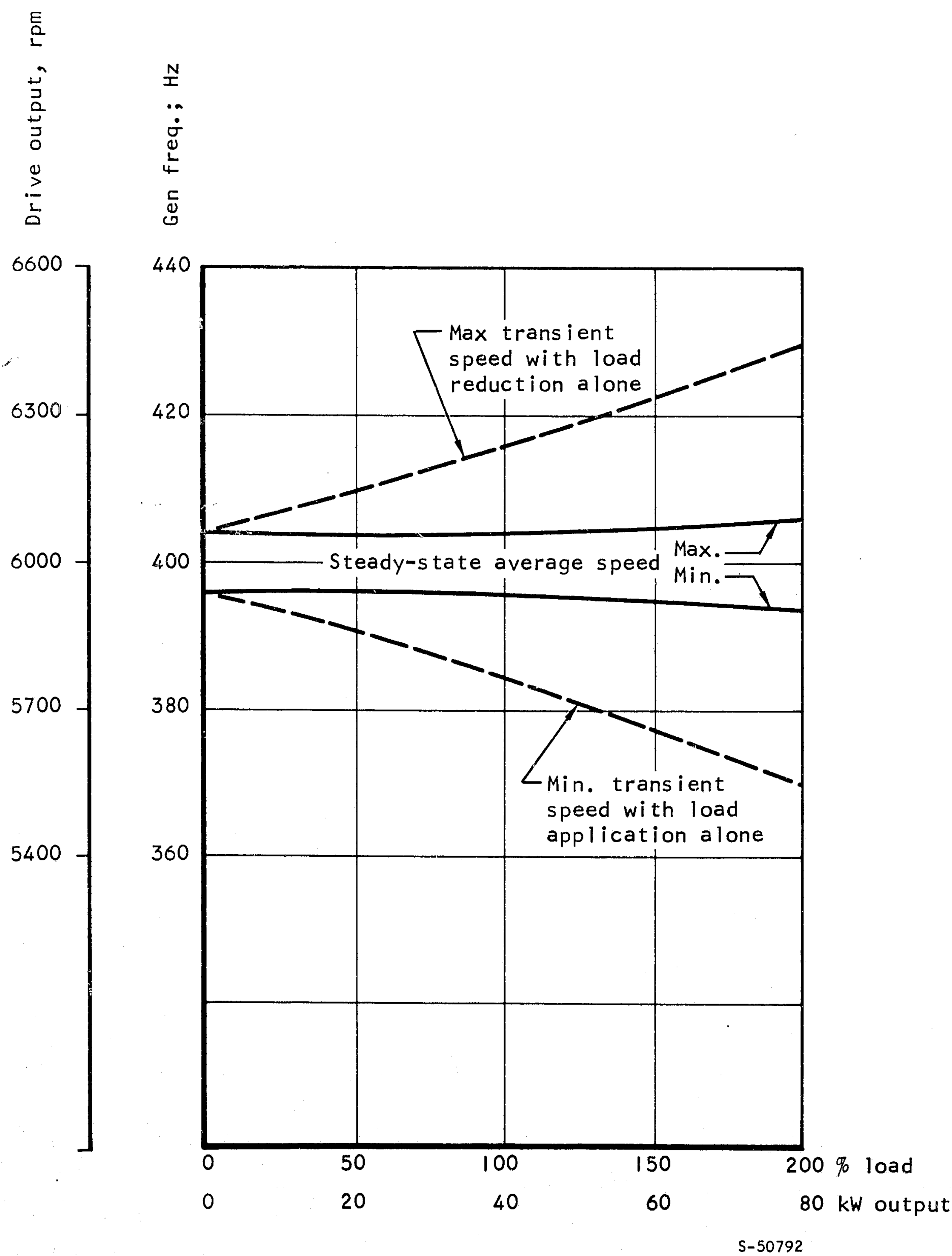


Figure A-8. Speed Load Characteristics

MIL-L-7808D Lubricating oil, gas turbine aircraft

MIL-C-8188C Corrosion preventive oil gas turbine aircraft.

MIL-I-6181D Interference control requirements, aircraft equipment.

The total generating plane comprises four generating channels driven by the main aircraft engines and one generating unit driven by an APU. No damage or degradation of performance will result from improper operation of the auto-paralleling circuits if they permit random paralleling to occur.

A load controller supplies a trimming signal to each drive governor to maintain load division within specified limits. Provision is made for a frequency reference to be added later that will hold frequency to limits as close as ±1/10 of 1 percent.

The following devices are provided:

(1) Shear section--A shear section is included in the input shaft of the drive with a shear torque of 6,000 ±450 in. lb. After a shaft shear, no harmful shipping of the sheared ends or damage to the engine output spline occurs.

(2) Overspeed--An overspeed device is built into the drive which operates at a setting corresponding to 465 ±15 Hz. This device independent of the main governor, positively shifts the drive to minimum speed ratio without external electric power to the drive.

The overspeed device automatically resets for normal operation when the drive is stopped.

(3) Underspeed--An underspeed device incorporated in the drive operates a single-pole, double throw, hermetically sealed switch with contacts capable of handling a control relay coil load with a nominal 30-V rectified power supply. The switch closes its normally open contacts when the drive output speed increases to correspond to 380 ± 8 Hz. These contacts reopen when the drive output speed drops to 360 ±8 Hz. Current is from .05 to 1 amp.

(4) Overrunning--An unloading device is proved to permit the generator to rotate more rapidly than the drive. When the generator is overrunning, the drive input reverse torque will not exceed 10 percent of the torque at rated load at minimum rated input speed.

(5) Filter--A 10-m oil filter and bypass valve are incorporated in this unit for protection against contaminated oil.

(6) Malfunction detection--Typically, three malfunction detecting devices are included:

(a) Temperature detecting devices to indicate rise in temperature of oil passing through the drive, and by switching, temperature of oil leaving the drive, or a device to power an indicating light when sump oil reaches excessive temperature.

(b) A switch to light a 28-V incandescent lamp when charge oil pressure drops below safe limits if charge oil pressure is essential to prevent damage to the drive.

(c) A magnetic/electric sump plug installed to permit a ground check for magnetic particles. The plug is designed so that the particles can be inspected without draining the sump.

(7) Disconnect--The input disconnect device is actuated by an electric solenoid. Maximum solenoid power is 5 A/.2 sec at 18 to 30 V dc. The disconnect can be operated at any engine operating speed and generator load; completes the disconnection within 3 sec after actuation. Continued operation of the engine after disconnect is not affected. Re-engagement of the disconnect is accomplished manually on the ground without any disassembly or dismounting of the drive. Disconnection does not occur without solenoid actuation; after disconnection has been effected, inadvertent re-engagement does not occur inadvertently.

Heat produced by the drive is transferred into the circulating fluid which is cooled in a separate oil cooler. Oil and airflow to the oil cooler does not exceed 11 gpm. The minimum efficiency of the drive at rated load is not less than 80 percent over the rated speed range. Maximum pad temperature is 350°F.

Generator Requirements

Design.-- The generator does not utilize commutators, slip rings, or brushes. Field flusing is not required for generator build-up. The regulator is a static type of design capable of supplying the necessary excitation to the generator during all the specified operating conditions. No magnesium parts are used in the generator.

Generator rating.-- The generator is rated 40 kVA with a power factor from 1.0 to .75 lagging. Power to operate the control panel and all contactors within the PMG generating channel are supplied by the PMG.

The generator and voltage regulator are capable of delivering the following loads at the conditions specified in MIL-G-6099, para. 4.5.4.

(1) 150-percent rated current at minimum rated lagging power factor for 5 min.

(2) 200-percent rated current at minimum rated lagging power factor for 5 sec.

(3) 125-percent rated current at minimum rated lagging power factor continuously from 71°C at sea level to -10°C at 50,000 ft.

The rated generator voltage is 120/208 V. The rated system voltage is nominally 115/100 V. Any variation from the nominal setting of 115/200 V will not exceed ±2.5 percent during steady state conditions at any load up to 50 kVA. These limits apply in lieu of the steady-state limits shown on fig. 8 of MIL-G-6099.

The generator is three phase, the frequency range is from 380 to 420 Hz, and the maximum speed for regulation is 6300 rpm. Efficiency at rated load is not less than 85 percent. The temperature-altitude range shall be Class C for the generator, and Class B for the regulator.

Any internal mechanical failure of the generator will provide a grounded circuit to light a warning light. The generator will not become a fire hazard and the voltage regulator will not be damaged.

The voltage regulator utilizes average sensing with highest phase limiting to a maximum of 125-V rms line-to-neutral. Voltage regulator sensing is at the load end of the generator feeders. The transient response of the generator and voltage regulator combination to a load disturbance is in accordance with the curves in fig. A-9. The output voltage remains within these limits rather than those of fig. 8 of MIL-G-6099.

Speed range.-- The speed range is from 5700 to 6300 rpm. Overspeed is 9500 rpm as required in the test requirement section of MIL-G-6099.

Generator weight, including the blast cap, does not exceed 90 lb; Regulator weight will not exceed 5 lb.

The generator and voltage regulator are in accordance with control drawing specified. The generator is 11.5 (dia) x 15 in., and the regulator is 6 x 6 x 5 in. maximum.

The overhung moment does not exceed 550 lb-in.

Generator Control Panel Requirements

Design and construction.-- The generator control panel weighs less than 12 lb. Dimensions are in accordance with ARINC size 1/2-ATR long.

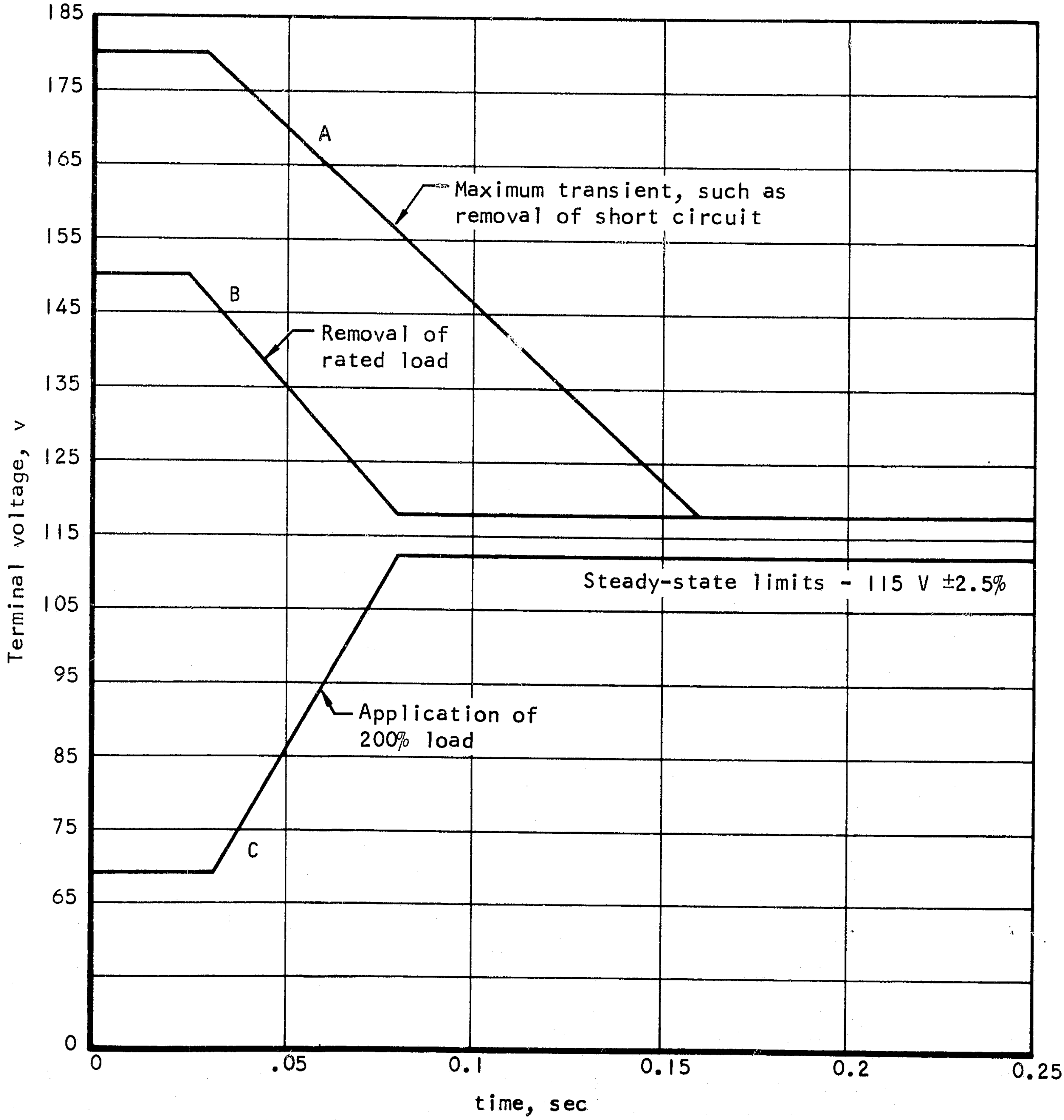


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Figure A-9. Transient and Steady State Voltage Limits

Performance.-- The panel shall be capable of operating satisfactorily under a temperature range in accordance with MIL-G-6099 fig. 2, Class B. It performs the following protective functions, without nuisance tripping:

(1) Overvoltage--The generator is deenergized and disconnected from the load bus before any phase voltage exceeds an inverse time curve of 5 V above 129 V up to an upper limit of 190 V.

(2) Overexcitation--The generator is selectively disconnected from its load bus and deenergized when it becomes over-excited in parallel operation. The load bus continues to be supplied from the tie bus without interruption. Protection is coordinated with the reactive load division circuit.

(3) Undervoltage--The generator is prevented from being connected to its load bus until all its phase voltages rise above 103 ±.3 V, and is disconnected from its load bus and deenergized when any phase voltage drops to and remains below 103 ±.3 V for the coordinated time delay period of approximately 6 sec. There is a minimum differential of 1 V between the pickup and dropout values.

(4) Underexcitation--During parallel operation the generator is selectively disconnected from its load bus and deenergized when it becomes under-excited for the coordinated time delay period of approximately 6 sec. The load bus continues to be supplied from the tie bus without interruption. When the undervoltage condition is not accomplished by absorbed reactive current, the tie bus contactor for the generator is opened after a coordinated time delay period of approximately 3 sec. Protection is coordinated with the reactive load division circuit.

(5) Underfrequency--The generator is prevented from being connected to its load bus until the drive underspeed switch closes; it is disconnected from its load bus when the drive underspeed switch opens for a period of 2 ± 1 sec. The protection is actuated from a normally open switch in the constant speed drive.

(6) Differential fault--A line-to-ground, line-to-line, or 3-phase fault on the generator feeders from the neutral side of the generator windings to the tie bus side of the bus tie contactors opens the holding circuit to the generator contactor coil and to the field relay coil within .03 sec after a fault current of .35 per unit or more starts to flow. If the fault current persists, the circuit to the holding coil of the associated bus tie contactor is opened after .4 ± .15 sec.

(7) Unbalanced current--A load division loop or other fault causing the phase current on one generator to differ by more than 25 A from the average of the generator currents of the same phase causes the associated bus tie contactor to open after a coordinated time delay of a nominal 8 sec.

(8) Anti-cycling--Neither the generator contactor nor the bus tie contactor shall close and reopen more than once on a fault condition while the controlling switches are maintained in the position to cause such closing.

(9) Diode failure--The generator is deenergized and disconnected from the bus in the event of diode failure in the generator field circuit.

Control functions.-- The generator control panel provides the following control functions:

(1) Exciter field control--Means are provided to deenergize the exciter field through operation of the generator control switch or any of the protective functions except underfrequency which cause the generator contactor to open. A latched relay is not used.

(2) Auto paralleling--Means are provided to close the contactor for a generator to be paralleled with others when it is within 90 electrical degrees of being in phase with the generator or generators connected to the tie bus. The control furnishes a closing signal to both the generator contactor and the bus tie contactor so that paralleling may take place across either. The paralleling transient does not exceed that of the switching 150 percent rated load.

(3) Sequential operation--System circuits cause each generator to take over its own load bus from external power when the generator is brought up to operating speed. A sequential selective circuit causes one generator only to be connected to the tie bus when external power is disconnected. The other operating generators are automatically parallel to it.

(4) Power supply--PMG power is converted to simultaneously supply the internal needs of the control panel, to energize the generator contactor, and to energize the essential bus contactor. The coil characteristics of the generator contactor and the essential bus contacare:

voltage range - 16 to 29 V, 28 V nominal; pickup voltage - 18 V maximum. The coil has a maximum inrush of 5 A and a hold current of .75 A.

(5) Extra circuits--Three sets of contacts are provided for the generator contactor and three are provided for the bus tie contactor to operate contactor/interlock circuits. These contacts are closed when the associated contactor is open. In addition, one set of contacts are provided to energize a warning light circuit when the generator is deenergized.

Bus Protection Panel Requirements

Design and construction.-- The bus protection panel weighs less than 6.5 lb and is in accordance with ARINC size 3/8-ATR short.

The panel is capable of operating satisfactorily under a temperature range in accordance with MIL-G-6099, fig. 2, Class B. The following functions with respect to external and auxiliary power control, are grouped, if necessary, in the bus protection panel. A three-position selector switch selects either external or auxiliary power.

(1) Connection--Circuits prevent either external or auxiliary power from being connected to the tie bus when the tie bus is powered from another source.

(2) Disconnection--A circuit is provided so that when the tie bus is powered from external or auxiliary power the bus tie contactor of a generator coming up to speed will be opened before the generator contactor closes.

(3) Power supply--A transformer rectifier utilizes power from the external ac power supply to supply 28 V dc power to close and hold the external ac power contactors.

(4) Phase sequence protection--The external power contactor is prevented from closing when the phase sequence is incorrect.

(5) Undervoltage--A means is provided to open the external power contactor and prevent cycling when any phase voltage drops to and remains below 90 V for 6 sec, regardless of whether the undervoltage is due to a fault on the tie bus, or low voltage from the external power supply. The external power contactor does not close if the supply voltage is below 100 V.

APPENDIX B

SAMPLE SST ELECTRICAL LOAD DATA

The following load data, load profile, and distribution schematic (for a sample supersonic transport) are included here for the purpose of comparison with the electrical system data of subsonic aircraft presented earlier in this report. The navigation and communication power supply data, furnished by the Collins Radio Company, are a preliminary configuration designed solely to obtain an estimate of the power loading on a sample supersonic transport aircraft.

TABLE B-1

SAMPLE SST AC LOAD ANALYSIS
LOADS IN KVA

Item	Conn load	Engine start	Taxi	Takeoff and climb	Super-sonic cruise	Sub-sonic cruise	Holding pattern and land-ing	Roll-out	Taxi
Lighting	25.2	13.3	14.2	15.0	13.0	14.0	15.9	15.9	14.0
Electronics	7.3	3.5	3.6	4.0	4.0	3.0	2.9	2.9	3.5
Trans/rect unit	6.4	1.3	2.0	2.1	2.6	2.4	2.1	2.1	2.1
MUX, EI, throttle, misc	9.5	7.7	6.5	7.1	7.1	6.7	6.5	6.5	6.5
Lavatories	5.2	.2	.2	.2	2.2	2.2	.4	.4	.4
Elec. equip, Racks and ADS cool	14.3	14.3	14.3	14.3	----	----	14.3	14.3	14.3
Brake cool	16.0	16.0	----	----	----	----	----	----	----
Windshield heat	7.7	----	5.8	5.8	----	----	5.8	5.8	5.8
Fuel pumps	135.6	20.0	50.4	51.2	68.0	51.2	40.0	40.0	40.0
Galleys	20.4	20.0	20.0	20.0	20.0	20.0	5.0	5.0	5.0
Demand on power system (kVA)	----	92.3	112.4	116.2	112.0	98.1	91.7	91.7	98.2

Sum of connected loads $\doteq$ 243 kVA

TABLE B-2

SAMPLE SST DC LOAD ANALYSIS

	LOAD IN AMPERES								
Item	Conn load	Start	Taxi	Takeoff and climb	Super-sonic cruise	Sub-sonic cruise	Hold and land	Roll-out	Taxi
Lighting	7.8	7.8	7.8	7.8	7.8	7.8	7.8	7.8	7.8
Electronics	90.6	31.4	34.3	38.6	49.3	49.3	38.5	38.5	38.5
Misc	3.6	3.6	3.8	3.6	3.6	3.6	3.6	3.6	3.6
Flight control	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
Fuel indication	58.2	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.5
Totals	170.2	54.3	57.2	61.5	72.2	72.2	61.4	61.4	61.4

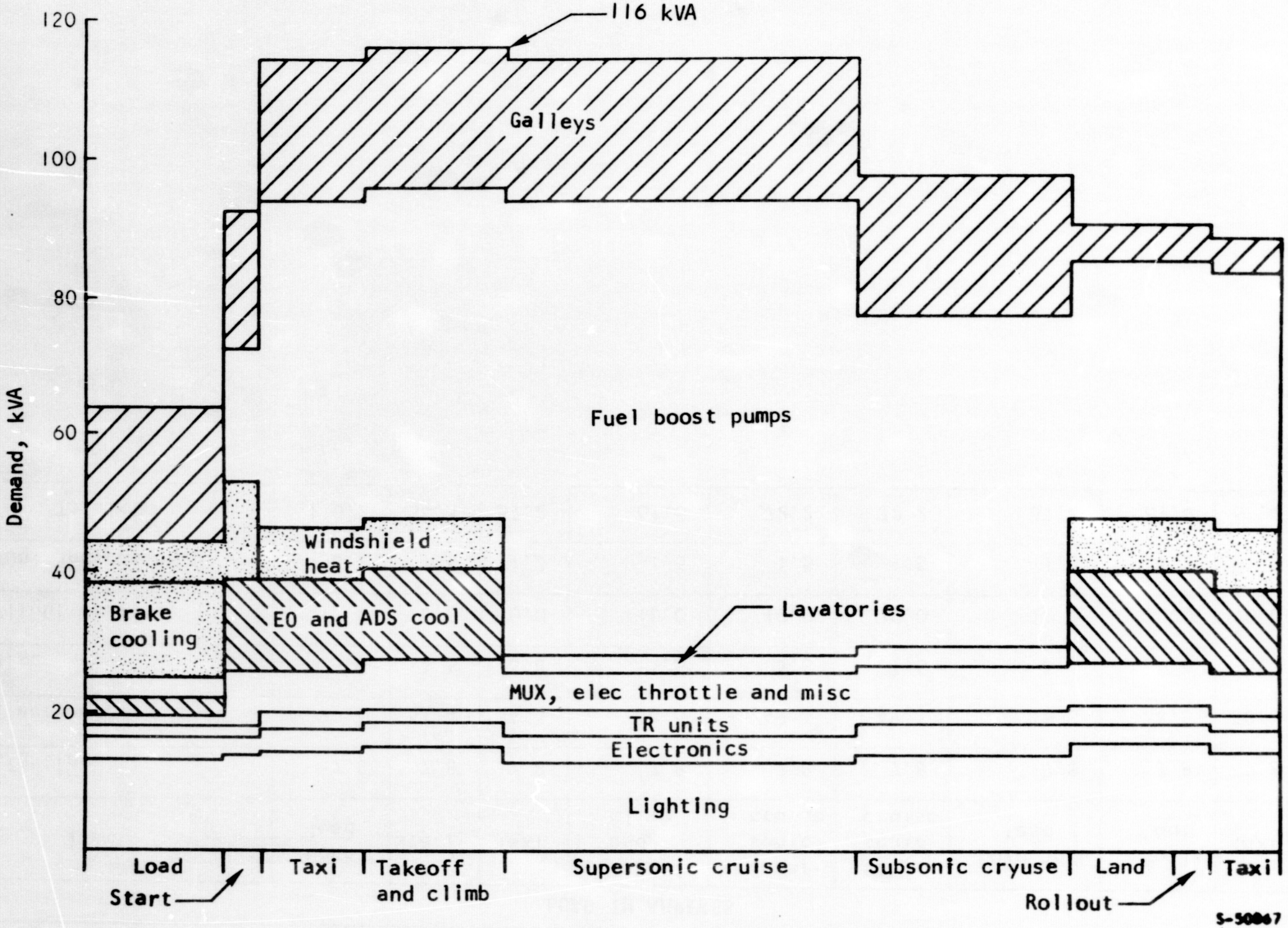

Figure B-1. Sample SST AC Load Profile

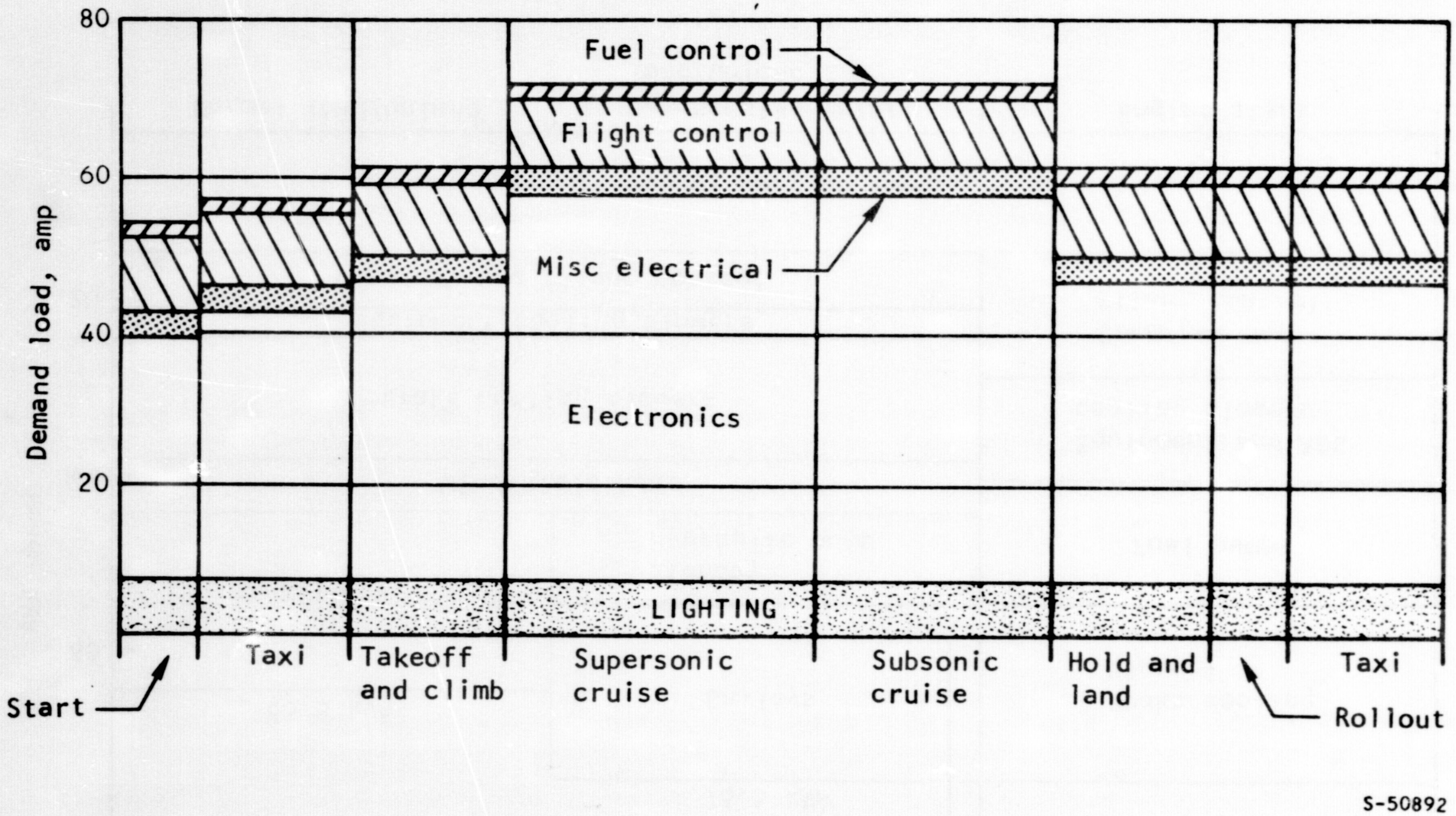

Figure B-2. Sample SST DC Load Profile

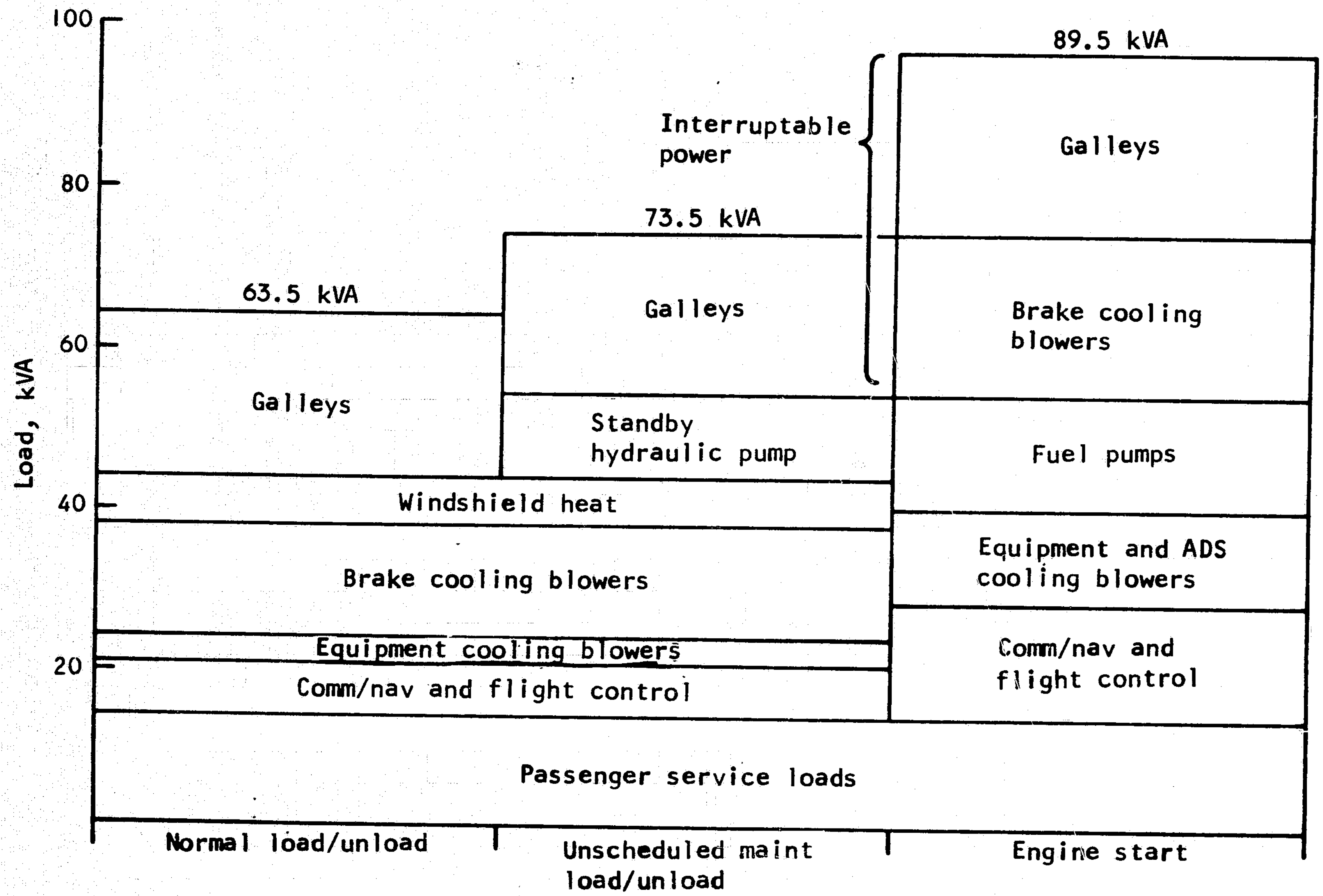

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Figure B-3. Sample SST Ground Power Loads

TABLE B-3

SAMPLE SST STANDBY POWER LOADS

Load	Battery, amps
Emergency lighting	4.0
Electronics	6.0
Flight instrumentation	4.5
Flight control	7.7
Flight safety	2.0
DC/AC inverters	92.0
30-min load	116.2

TABLE B-4

SAMPLE SST STANDBY POWER SYSTEM INVERTER LOADS

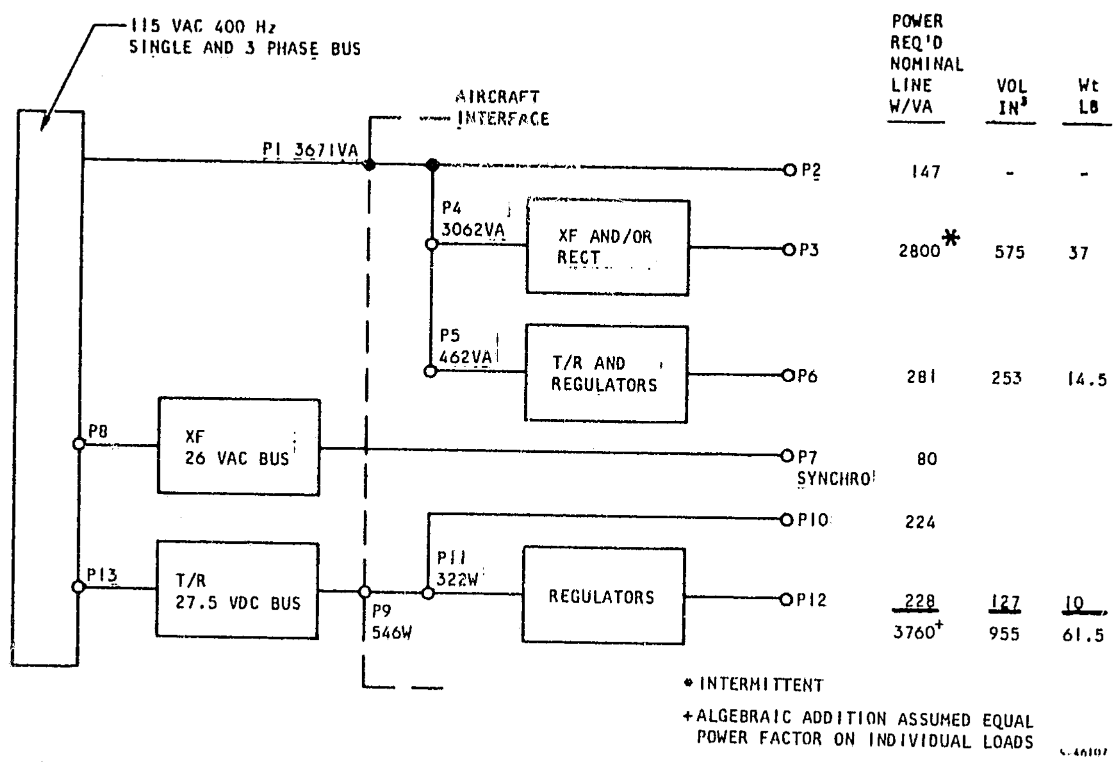
Load	Output, volt-amps
Electronics	310
Flight instrumentation	285
Flight control	1,205
Total output	1,800 va at .96 PF

Inverter input at 26 vdc 75-percent efficiency - 92 amps

TABLE B-5

SAMPLE SST VSCF CHANNEL RATING

Channel Rating	10,000 to 15,000 rpm	15,000 to 20,000 rpm
Continuous	80 kVA	75 kVA
10 min	75 kVA	75 kVA
5 min	90 kVA	90 kVA
1 per unit-base rating -- 60 kVA		



Unit	P2	P3	P4	P5	P6	P7	P9	P10	P11	P12	Volume, cu in.			Weight, lb		
	va	w	va	va	w	va	w	w	w	w	P3	P6	P12	P3	P6	P12
ADF 51Y-7						5	35		33	21			25			1.5
XPONDER 621A-6		22	28	45	18						4	21		1	3	
VOR/LOC/ES 51RV-2B						10	44		44	28			25			1
DME 860E-3		10	11	79	44						3	47		0.5	2.5	
VHF COMM 618M-2B							193		193'	144			60			3.5
RAD ALT 860 F-1		26	27	64	38						4	43		0.5	2.5	
MKR BEA 51Z-4							6		6	2			2			1
PASS AD AMP 346-1B							225	192	33	25			10			1
VOICE REC 642C-1	22			19	15							8				
HF 718T	95 3∅	1151	1230	233	166						200	134		11	6.5	
SEL CAL 456C-1	15	15	15				27	21			3			1		
AIR DATA COMP	30	25	28			65					15			2		
SAT COMM PRE AMP 522C-1							10		10	6			3			1
SAT COMM PA 548Y-1	3∅	1507	1675								334			17.5		
MAG COM MC102		44	49								12			3.5		
INTERCOMM 346B-3							14	11	3	2			2			1
TOTAL	147	2800	3062	462	281	80	546	224	322	228	575	253	127	37	14.5	10

Figure B-4. Sample SST Navigation and Communication Gear Power Supply Data

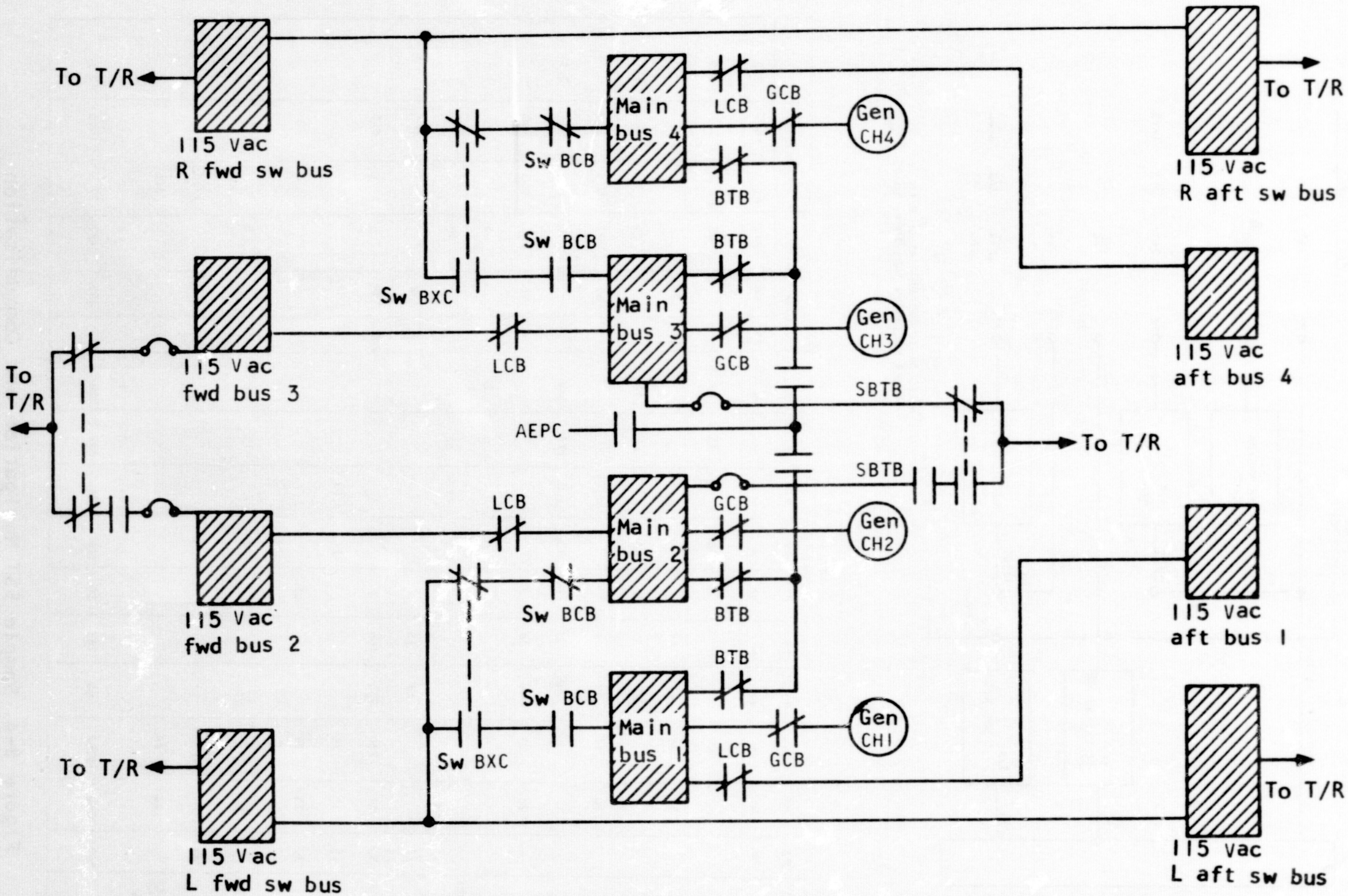


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Figure B-5. Sample SST AC Power Distribution Schematic

APPENDIX C

AIRCRAFT HYDRAULIC SYSTEMS AND EQUIPMENT

Hydraulic Systems in Existing Airplanes and/or Airplanes Under Construction

Background.--In older aircrafts, cables and other mechanical linkages are used to connect the flight controls with the aerodynamic control surfaces. Because of the high forces required to actuate the control surfaces, it has been necessary to incorporate hydraulic boost provisions in flight control systems for large high-performance aircraft. In addition, hydraulic power is used for other high-power intermittent services such as actuating thrust reversers, retracting landing gear, and steering the nosewheel during taxi. Because of the criticality of many of these functions, considerable redundancy is provided in the form of parallel hydraulic circuits, duplicate components, and other backup provisions. In some cases, direct mechanical operation serves as a backup or emergency mode. In others, the pilot is unable to provide the necessary control forces and backup operation must be provided in other ways.

As the power requirements increase, it has been necessary to provide greater redundancy in the hydraulic system itself. For example, many aircraft use two hydraulic systems with an emergency flight control capability provided by direct mechanical actuation through the control cables. Large transport aircraft, such as the DC-10, L-1011, B-747, or C-5, use three or four independent hydraulic circuits, each provided with more than one power source.

Design of these aircraft hydraulic systems is based upon reliability considerations for operation under various possible failure modes in the hydraulic system, in the actuation components, and in the system power sources. In most systems, engine-driven pumps serve as the primary source of power. Other power sources (used to supply backup or emergency hydraulic power) are (1) air turbine motors using bleed air from the propulsion engines, (2) gas turbine auxiliary power units, (3) electric motors, and (4) ram air turbines. These are used in different combinations, depending upon the application to specific design requirements, which vary considerably with the type of aircraft and its intended usage.

As indicated previously and shown in table C-1, one of the conspicuous developments with increasing aircraft size and performance has been the increase in hydraulic system power capability. The high hydraulic power capability shown for the more advanced systems reflects both the increased hydraulic power demand and the increased redundancy required to provide the desired reliability with a more complex system.

Hydraulic power schedule.-- Peak hydraulic power requirements are ordinarily attained during aircraft climb and approach for landing. As shown in the typical hydraulic load profile of fig. C-1, a sizable portion of the peak requirements are associated with operation of the landing gear and high-lift aerodynamic surfaces (slats and flaps). In addition, because of the requirements for flight

TABLE C-1

TYPICAL AIRCRAFT HYDRAULIC POWER CAPABILITY

Airplane	Hydraulic hp
DC-3	6
DC-4	34
DC-6	36
DC-7	37
B-737	73
DC-8	84
CV-990	168
DC-10	315
L-1011	385
B-747	500
C-5	850

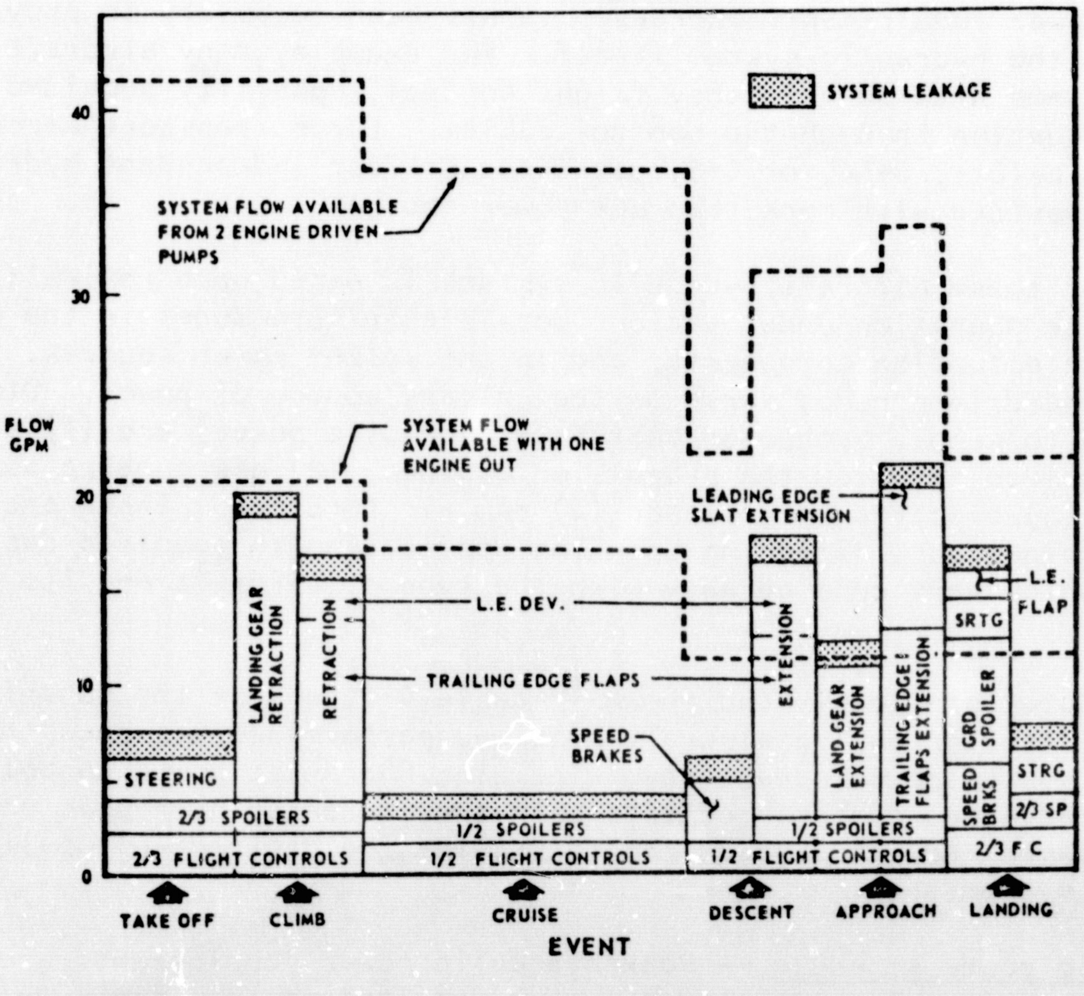


Figure C-1. 737 Hydraulic System Flow Capabilities and Demands (Ref. 41)

control system reliability, considerable redundancy is provided. This results in a substantial margin in available power over that required to meet the basic hydraulic power demand. Also, it will be noted that the hydraulic power requirement for the cruise condition will be substantially lower than the peak. Consequently, excess hydraulic power is available for auxiliary services during cruise.

Hydraulic system installation.--One of the most significant aspects of hydraulic system design is the means of transmitting the power from where it is generated (usually the engine driven pumps) to where it is used (primarily wings, empennage, and landing gear). As shown in table C-2, present transport aircraft utilize large amounts of tubing and fittings. These fittings have been troublesome in service and there is a definite trend in system design to reduce the number of threaded connections through use of permanent joints (welded, brazed, and permanently swaged fittings).

TABLE C-2

REPRESENTATIVE HYDRAULIC LINES AND FITTINGS

Airplane	Total length of hydraulic lines	Total number of fittings
B-707	2600 ft (790 m)	2500
B-727	3280 ft (1000 m)	2600
B-737	1542 ft (470 m)	970
B-747	3388 ft (1030 m)	1270

Several aircraft hydraulic systems, representative of contemporary design, are described below to illustrate the state of the art.

Hydraulic system in the Boeing 737 airplane.--Three hydraulic circuits are used in the B-737 airplane (fig. C-2), one serves as the primary system using engine driven pumps, and the other two serve as a backup system using electric motor driven pumps. The primary system has two 21-gpm (5.55 m^3/hr) variable-delivery pumps with one on each engine. In this way, full capability is provided with one engine out. The backup hydraulic system uses ac electric motor driven 6-gpm pumps (1.6 m^3/hr) to provide emergency hydraulic power for operation of the flight controls in case both engines are lost. The standby hydraulic system operates the essential flight controls for emergency operation.

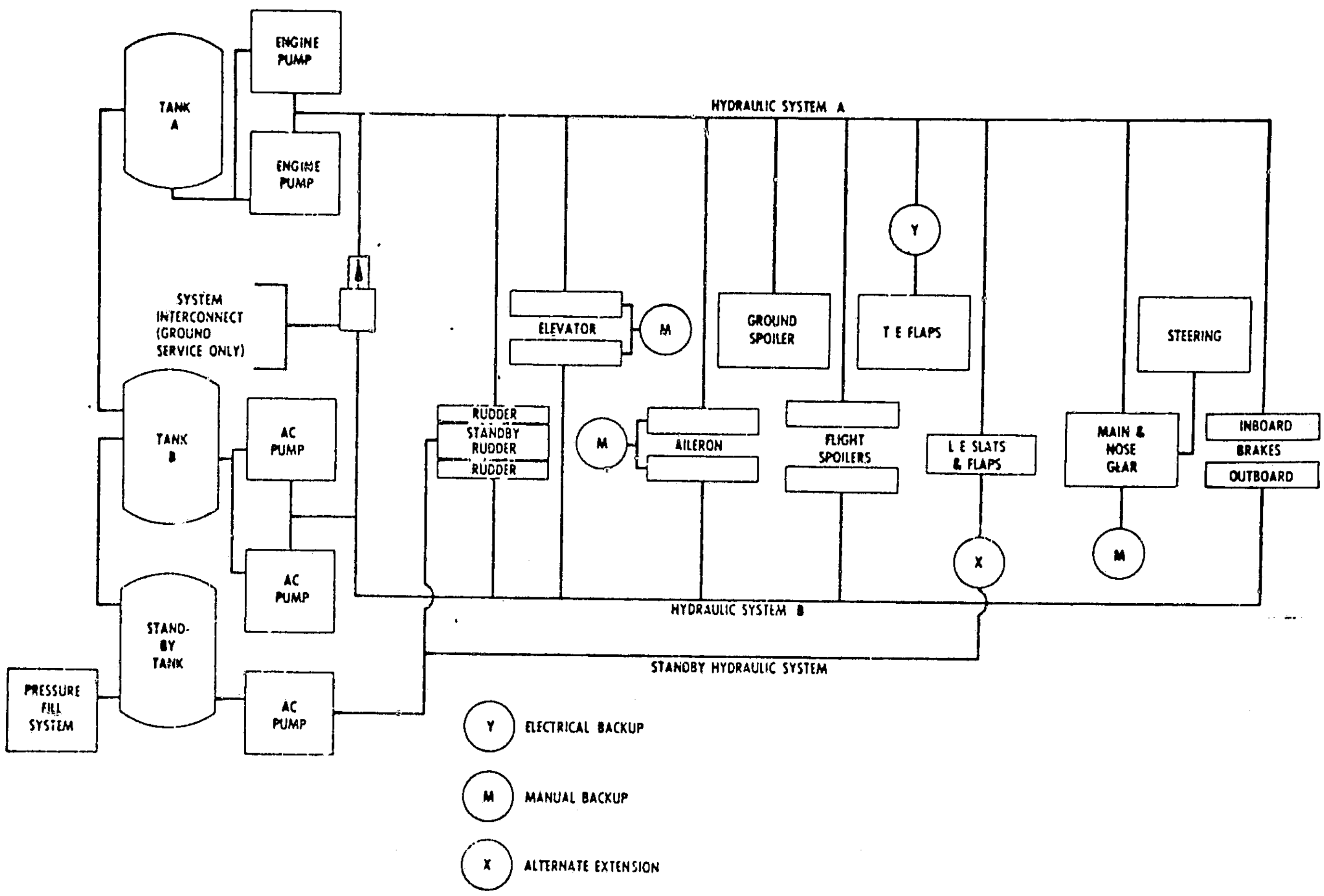


Figure C-2. 737 Hydraulic System Diagram (Ref. 42)

Hydraulic systems in the Boeing 747 airplane.-- Four hydraulic systems are used in the B-747 airplane, with the functional assignments shown in table C-3. Systems 2 and 3 are mainly reserved for flight controls; systems 1 and 4 are considered the auxiliary power systems. Each system shown in fig. C-3 has an engine driven pump rated at 38-gpm (10 m^3/hr) and an auxiliary pump (which is driven pneumatically with bleed air by an air turbine power unit) rated at 32 gpm (8.4 m^3/hr). The auxiliary pump is automatically put into operation if the engine fails or the engine driven pump does not supply the hydraulic demand.

Hydraulic system in the McDonnell Douglas DC-10 airplane.-- The DC-10 airplane uses three hydraulic systems, each with two engine-driven hydraulic pumps of 35-gpm (9.3 m^3/hr) capacity. Fig. F-4 shows the system functional block diagram. Two ac electric motor driven pumps (6-gpm, 1.6 m^3/hr), variable delivery) supply emergency and ground checkout hydraulic power. The gas turbine APU drives a 35-gpm (9.3 m^3/hr) pump as a standby source of hydraulic power. In addition, two power transfer motor-pump units permit transfer of power without interchange of hydraulic fluid from one system to another. Consequently, all three hydraulic systems remain operative in event of an engine failure. A ram air turbine supplies ac electrical power for emergency power if all engines are lost.

TABLE C-3

HYDRAULIC SYSTEMS FUNCTIONAL ASSIGNMENTS

System 1	System 2	System 3	System 4
Left outboard aileron	Left outboard aileron	Right outboard aileron	Right outboard aileron
Left inboard aileron	Right inboard aileron	Left inboard aileron	Right inboard aileron
Left outboart elevator	Left outboard elevator	Right outboard elevator	Right outboard elevator
Right inboard elevator	Right inboard elevator	Left inboard elevator	Left inboard elevator
Upper rudder	Lower rudder	Upper rudder	Lower rudder
Nose gear actuator	----	----	----
Nose gear steering	----	----	----
Inboard TE flaps	Stabilizer pitch trim	Stabilizer pitch trim	Outboard TE flaps
Aft main gear actuator	----	----	Forward main gear actuator
Normal brakes (secondary)	Reserve brakes	----	Normal brakes (primary)
	Spoilers 2, 3, 10, 11	Spoilers 1, 4, 9, 12	Spoilers 5, 6, 7, 8

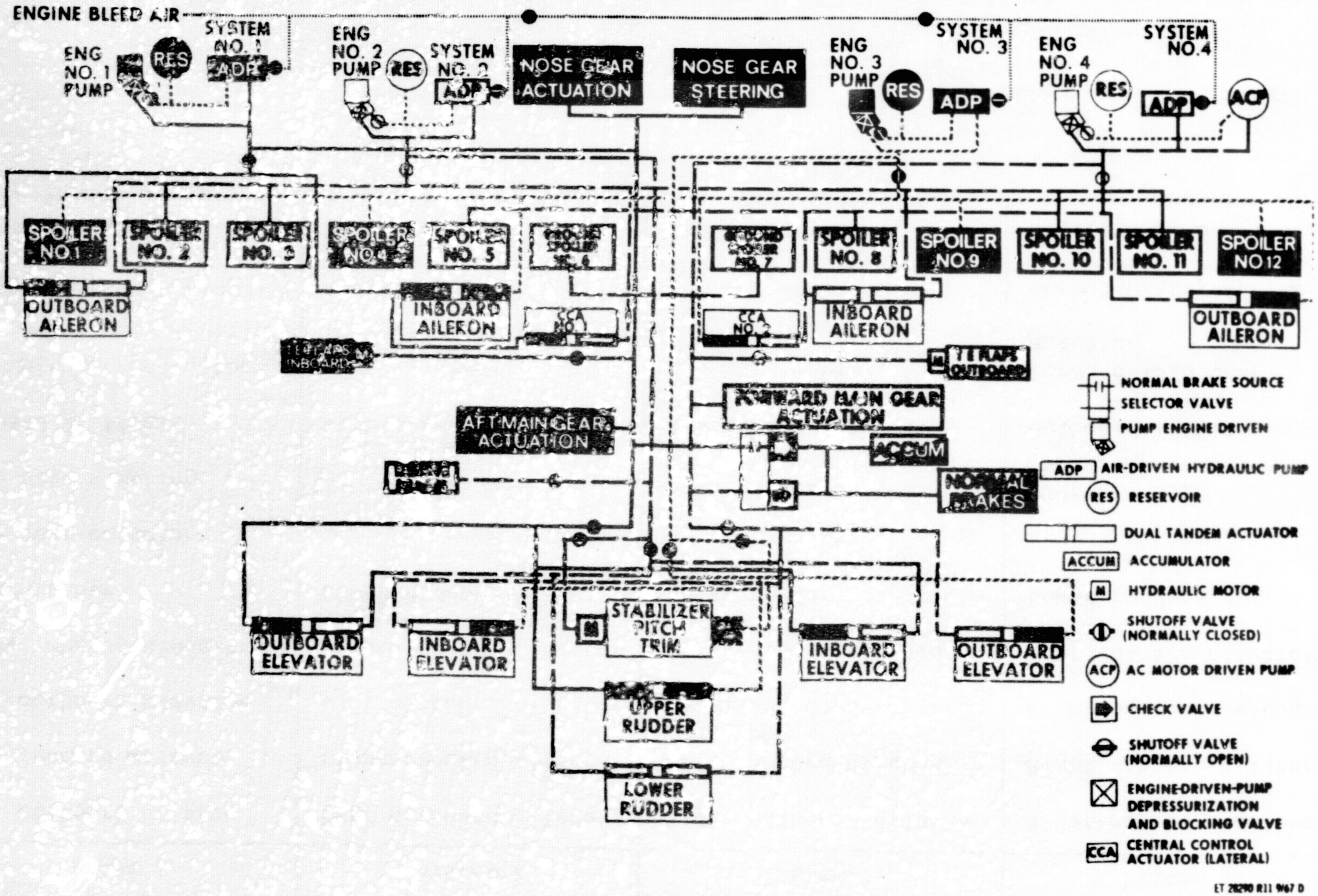
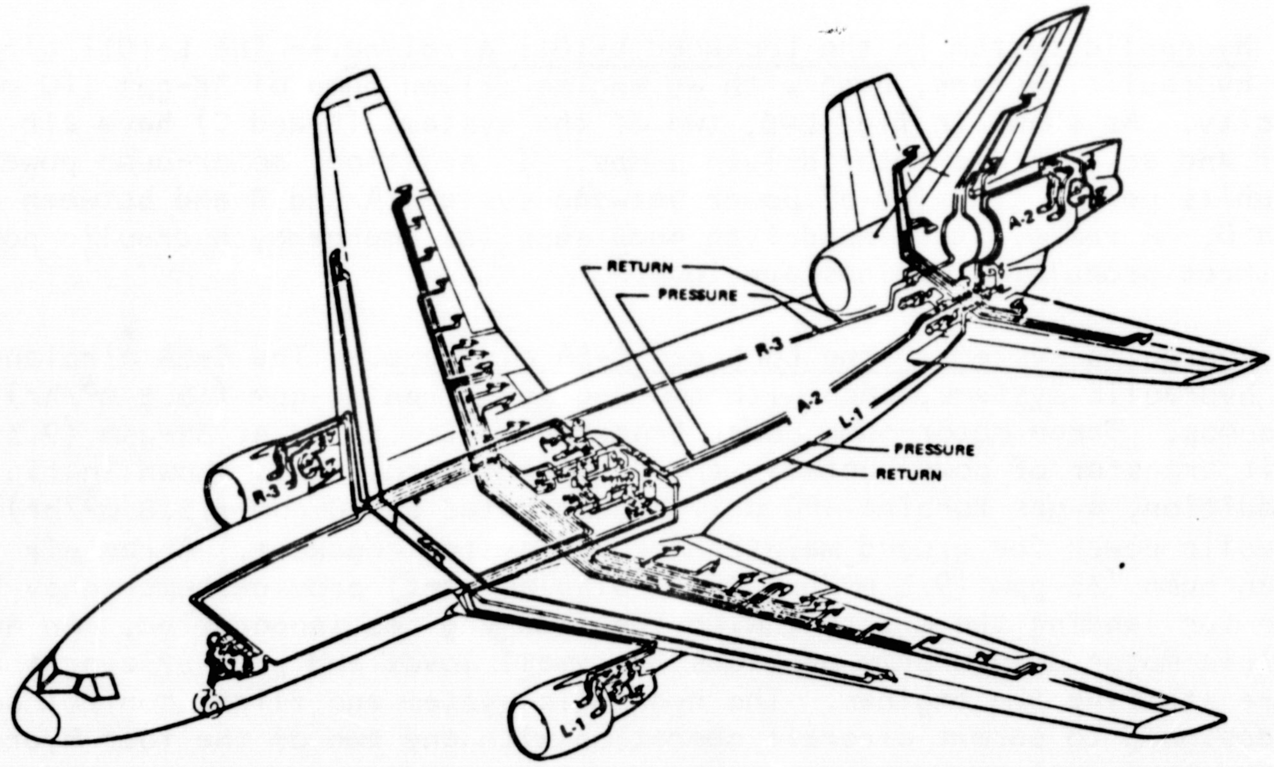
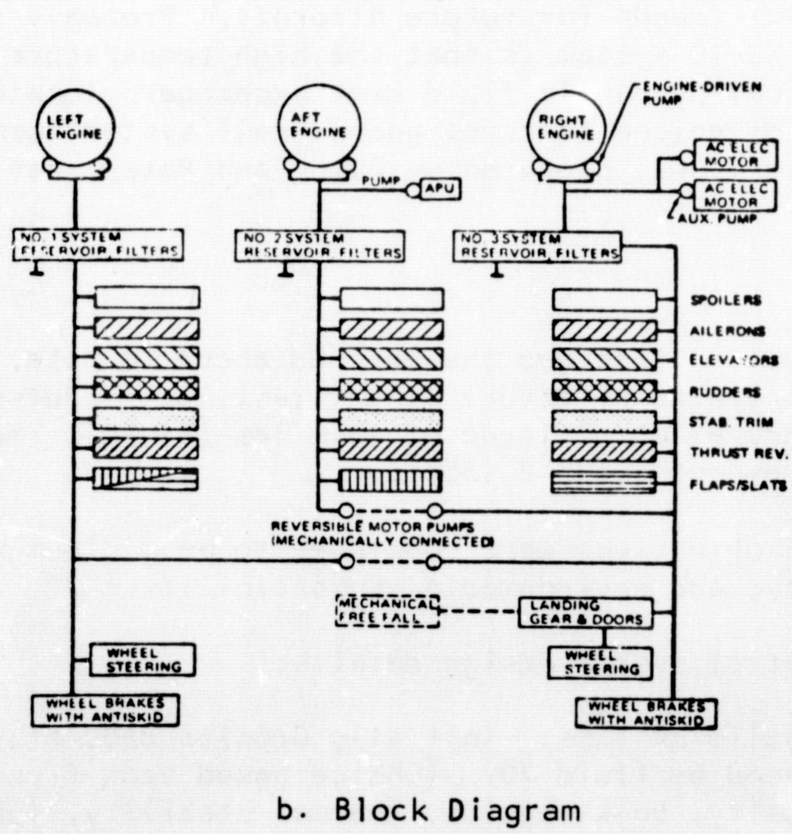


Figure C-3. 747 Hydraulic System Schematic (Ref. 43)



a. Aircraft Installation



b. Block Diagram

Figure C-4. DC-10 Hydraulic System (Ref. 44)

Hydraulic system in the Lockheed L-1011 airplane.-- The L-1011 airplane has four hydraulic systems, each with an engine driven pump of 38-gpm (10 m^3/hr) capacity. As shown in fig. C-5, two of the systems (B and C) have air turbine motor and ac electric motor driven pumps. In addition, motor-pump power transfer units permit transfer of power between systems A and B and between systems C and D. A ram air turbine driven pump supplies emergency hydraulic power if all three propulsion engines are lost.

Hydraulic system in the Lockheed C-5A airplane.-- The C-5A airplane uses four hydraulic systems, each with two engine-driven 60-gpm (15.8 m^3/hr) hydraulic pumps. Three motor-pump power transfer units, rated at 35-gpm (9.3 m^3/hr), permit transfer of power between the different circuits as shown in fig. C-6. In addition, a gas turbine APU driven pump rated at 60-gpm (15.8 m^3/hr) provides hydraulic power for ground maintenance and system checkout. A ram air turbine driven pump (35-gpm (9.3 m^3/hr) fixed displacement) provides emergency hydraulic power for landing the aircraft with the four engines inoperative. An additional electric motor driven pump provides for small loads and pressurizing the system before starting the engines. The hydraulic system and flight control actuators are designed to permit aircraft operation with any two of the four hydraulic systems inoperative.

Hydraulic systems in experimental and advanced aircraft.-- An indication of the state of the art is exemplified by the designs selected for the XB-70 and F-111. These systems indicate present optimization and system balance and reveal the development trends for future aircraft. Probably the most significant feature of the XB-70 system is that the high temperature exposure areas are fuel cooled in fuel-hydraulic fluid heat exchangers located in the system reservoirs. A list of unique features and overall system comments for the XB-70 and F-111, respectively, was prepared by Ramby and Patch, ref. 2, and is presented below.

XB-70

(1) Vehicle capabilities are the most advanced to date. These capabilities were demonstrated by actual flight test, with cruise speeds of 2000 mph (3220 km/hr) at an altitude of 13 miles (21 km). Maximum vehicle surface temperatures reach 675°F (357°C).

(2) The design objectives were concluded to be minimum weight, maximum reliability, and environmental compatibility.

(3) Flight control system design details:

(a) Hydraulic systems: initially Oronite 8200 fluid, later to be replaced by fluid 70. (Choice based upon five properties - viscosity, bulk modulus, thermal stability, lubricity, and weight.)

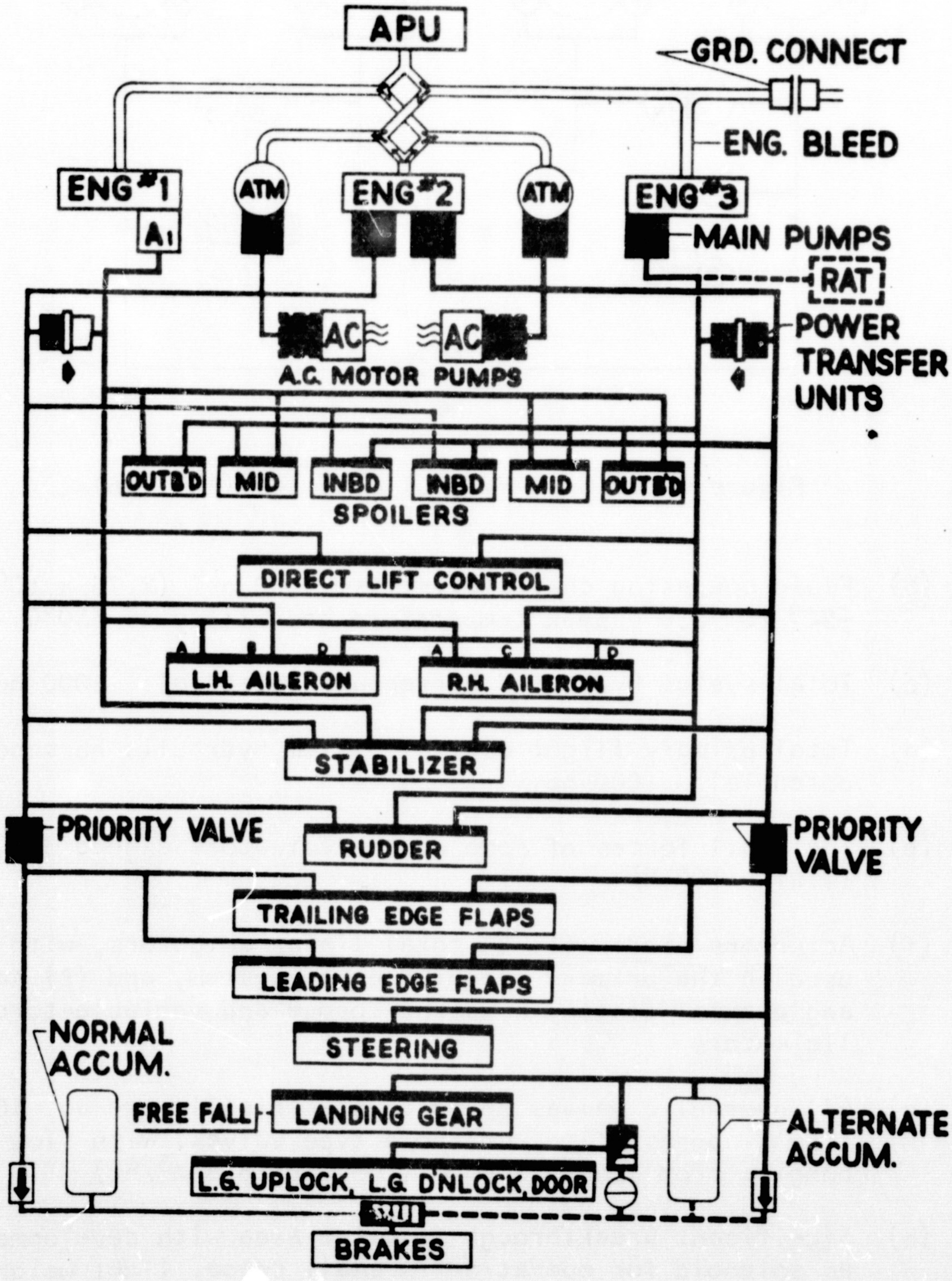


Figure C-5. L-1011 Hydraulic System (Ref. 45)

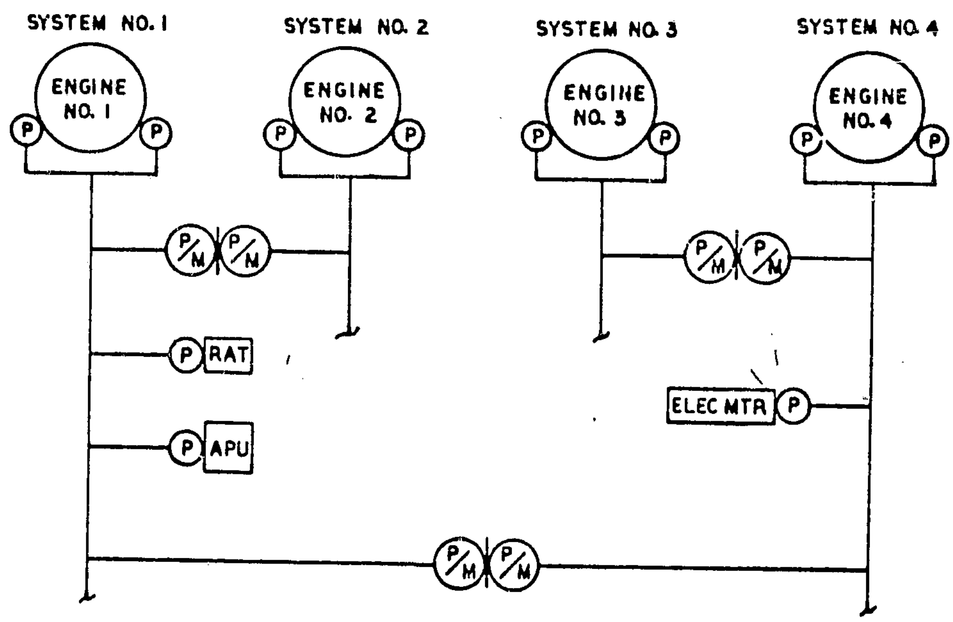


Figure C-6. C-5A Hydraulic System (Ref. 46)

(b) Fluid operating characteristics: 400 psi (2.75×10^6 N/m^2) - -54° to $+230^\circ$C, peak temperature capability of 330°C

(c) Total system hydraulic horsepower potential: 2000 hp

(d) Total primary flight control system hydraulic horsepower potential: 1000 hp

(e) The stall forces of some of the actuators exceed 316,000 lb (1.41×10^6 N)

(f) Actuators used: (1) 85 total linear actuators, with 39 being used in the primary flight control system, and (2) 44 fixed angle axial-piston, constant-torque and variable-torque hydraulic motors

(g) All hydraulic valves operated electrically are ac, 400-Hz, 115-V, nonrectified, solenoid type valves, with flow capacities ranging from .25 to 70-gpm (.066 to 18.5 m^3/hr)

(h) Significant breakthrough in valve area with development of pure ac solenoid for operation in small space, light weight, and high ambient (330°C) application

(i) Hydraulic fluid temperature is controlled by fluid-to-fuel heat exchangers located in the system reservoirs, heat exchanger flows are 650-lb/min (295 kg/min) fuel and 145-gpm (38.5 m^3/hr) primary flight control system fluid with a heat transfer of 19,500 Btu/min (34.3 kW)

(j) The development of a new pump/motor unit which is capable of being used as the engine starting motor and as the hydraulic power source

(k) The master-slave pump concept was used to decrease the self-generated heat within the system

(l) The greatest single development achievement was the successful design and testing of metallic seals

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(1) The most significant flight control specification is exterme reliability with fail operational capability.

(2) System specifications also require a high degree of resolution with low deadband characteristics.

(3) Flight control system design details:

(a) Hydraulic system MIL-H-5606 fluid

(b) First application of a hydraulic logic, failure correcting, redundant, series actuators

(c) Servovalves receive digital electrical signals directly from the PFCS computer

(d) Completely fly-by-wire wing spoiler controls

(e) Temperature specification: -54° to $135^{\circ}C$

Parametric Information of Current Aircraft Hydraulic Equipment

The components of a hydraulic system usually include a pump, fluid transmission lines, servovalves, actuators, and reservoirs and/or accumulators. System accessories include check valves, sumps, flow limiters, relief valves, service connections, filters, etc. For a flight control system, two additional components are included: control signal summers and power control devices. Hydraulic servovalves are usually the control devices.

Hydraulic Pumps.-- The source of hydraulic power is usually obtained from a fluid pump. Fluid pump designs include internal gear, external gear, balanced vane, unbalanced vane, and both fixed and variable displacement pumps of the inline-, bent-axis-, or radial piston types. Characteristics of major pump types are presented in table C-4 and fig. C-7.

TABLE C-4

CHARACTERISTICS OF MAJOR PUMP TYPES

Characteristics	Unbalanced vane	External gear	Swash-plate piston
Maximum pressures,* psi	250 to 1000	250 to 2500	1000 to 5000
Flow, gpm	.5 to 100	.2 to 150	.5 to 450
Operating speed,* rpm	1200 to 2000	1800 to 7000	Medium high

*For continuous system operation. Higher speeds apply to smaller units. Maximum speed generally decreases with increased displacement.

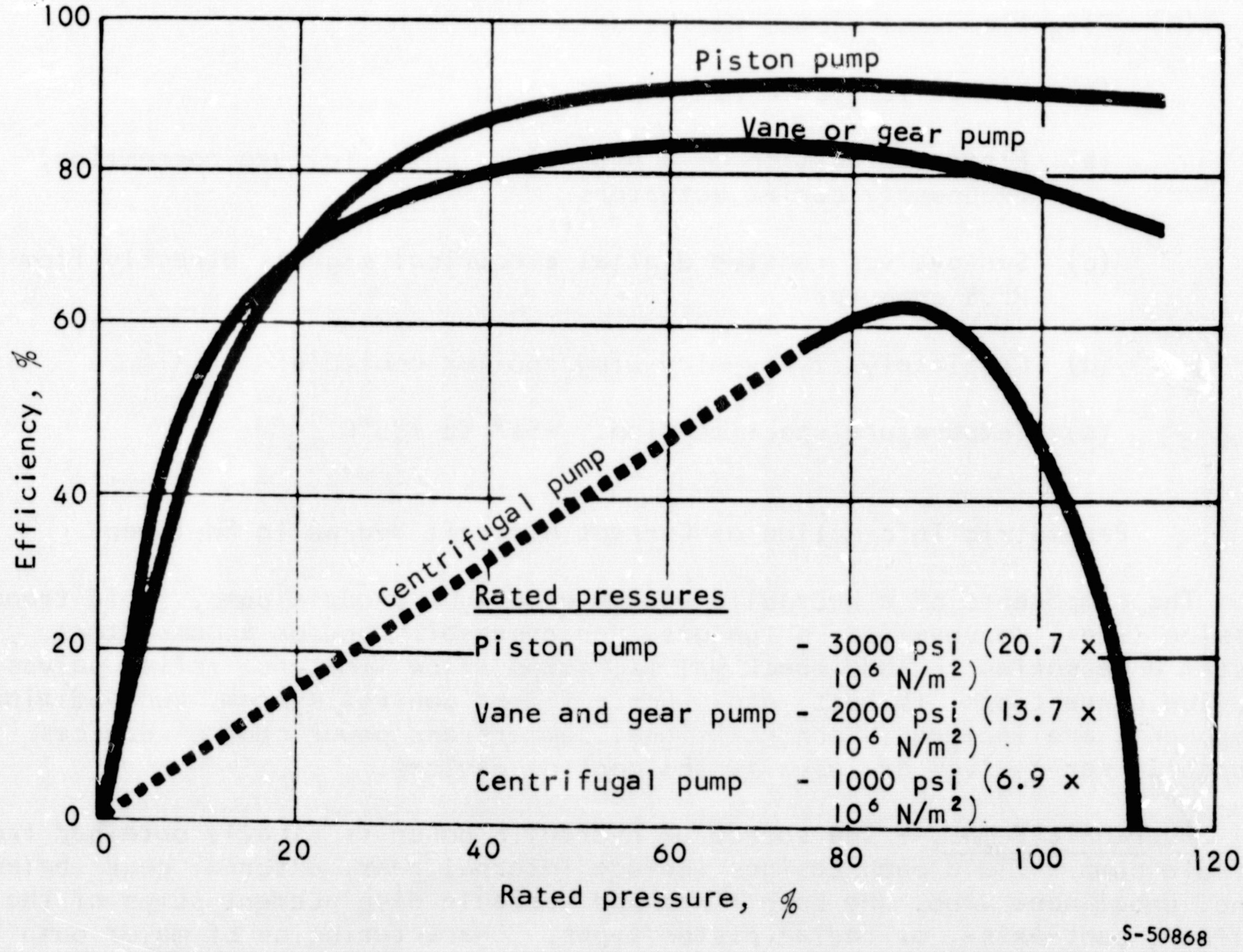

Figure C-7. Efficiency Characteristics for Typical Pumps (Ref. 47)

According to the survey conducted by Ramby and Patch (ref. 2), the most common aircraft main hydraulic power source in use today is a pressure compensated, variable displacement, axial piston pump which delivers constant pressure and variable flow hydraulic power. The reasons given for the wide usage result from:

(1) Use of multi-pump configurations for weight and space conservation, and increased reliability.

(2) Reduced sensitivity of system pressure and flow characteristics with respect to variations in engine speed.

(3) Use of constant speed drive to maintain constant frequency in electrical generators.

(4) Design compatibility that results in design simplification when several functions are being powered by the same hydraulic system.

These pumps have low inertia, thus affording a rapid start/stop capability and high horsepower per unit volume and weight.

The pump is available in both the inline and bent-axis configurations for both the fixed and variable displacement designs. Fig. C-8 presents the rotational speed range as a function of the hydraulic horsepower for both designs, as calculated from Vickers catalog data. For the same series of pumps, fig. C-9 presents the weight based on the rated speed.

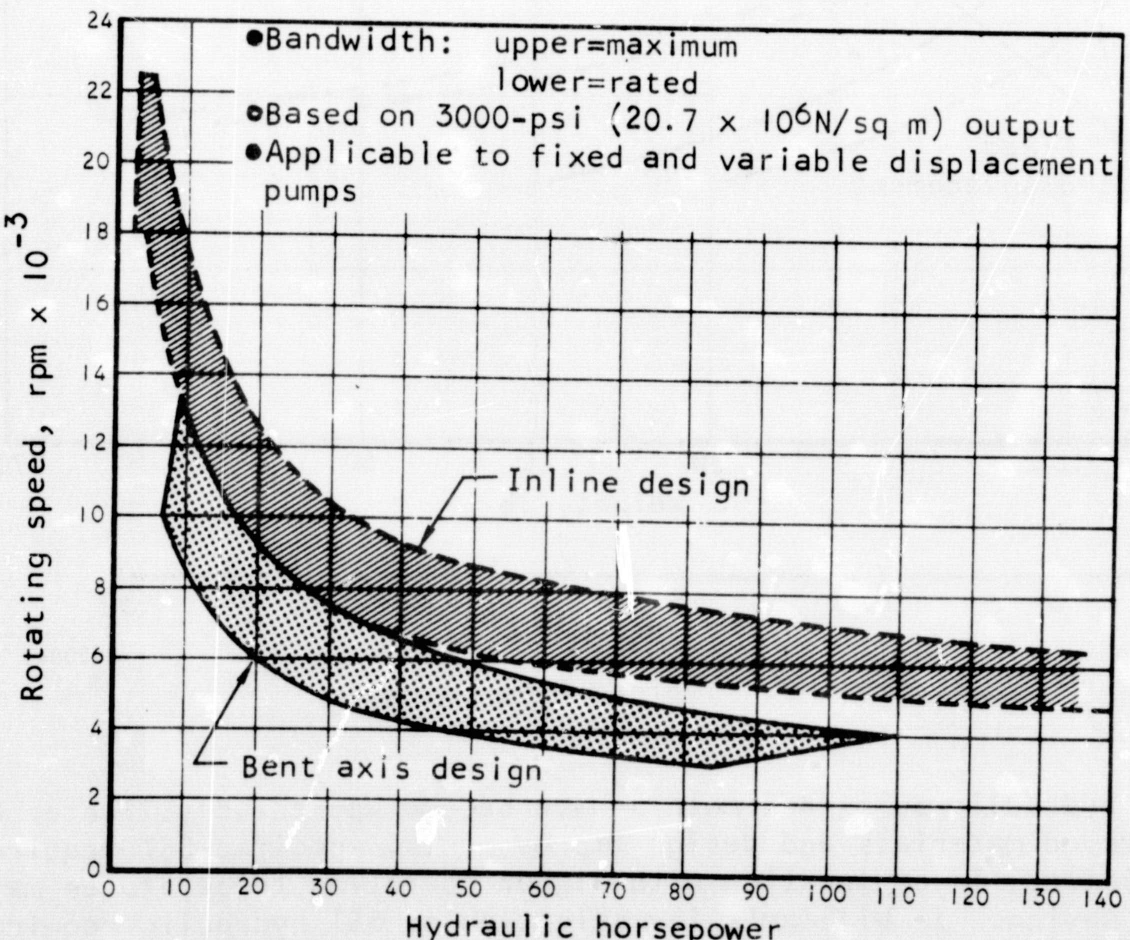

Figure C-8. Axial Piston Pump Operating Speed Ranges

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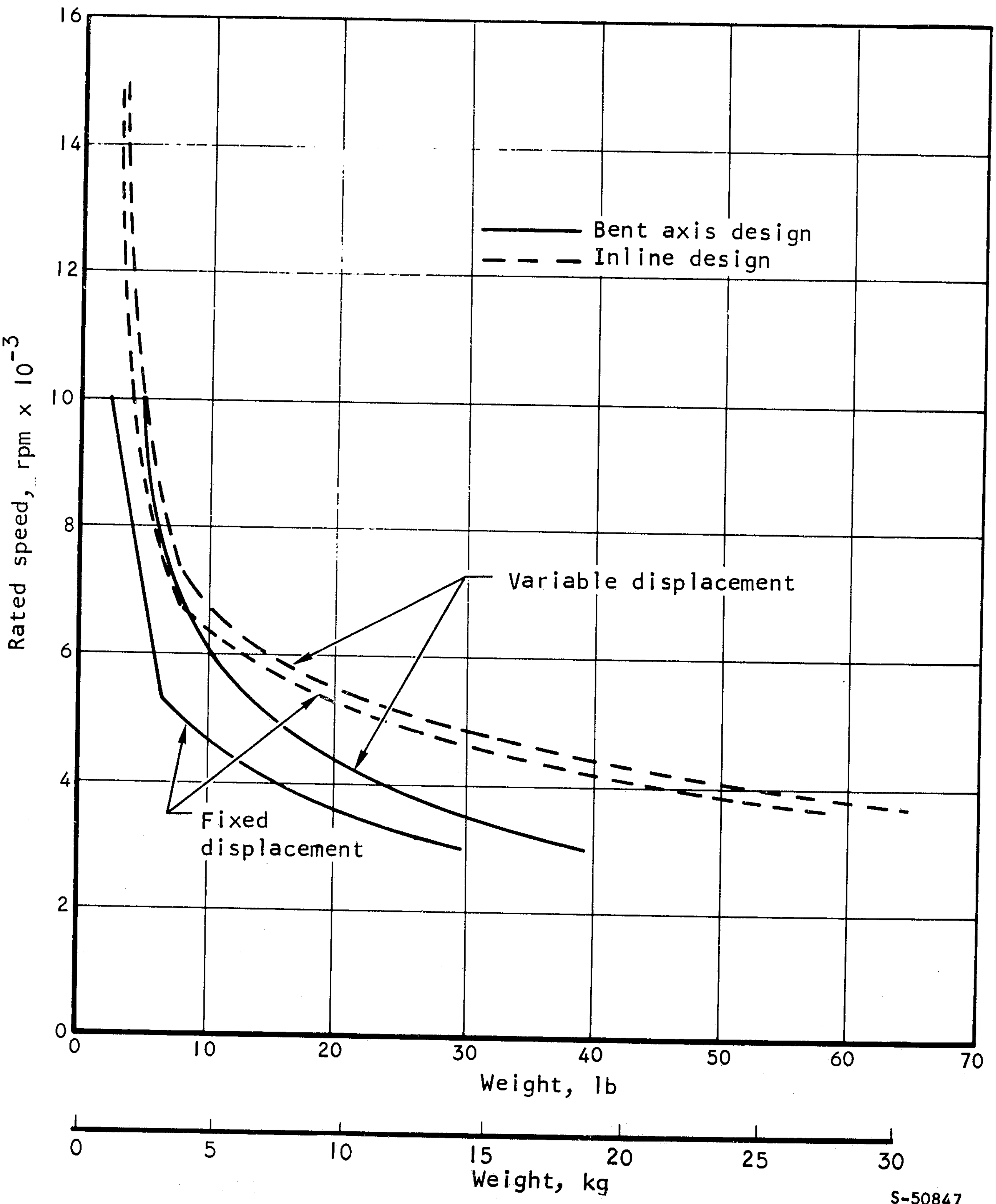

Figure C-9. Typical Pump Weights

Although hydraulic pumps and motors have become appreciably lighter as a result of improved materials and design improvements, specific SST requirements of higher efficiency in conjunction with higher operating temperatures may offset the potential saving. In William's investigation of SST hydraulic requirements (ref. 48), comparison of existing and estimated motors and pumps was compiled as shown in fig. C-10. The development effort suggested is a renewed effort to lower the specific weight while maintaining the higher operational temperature.

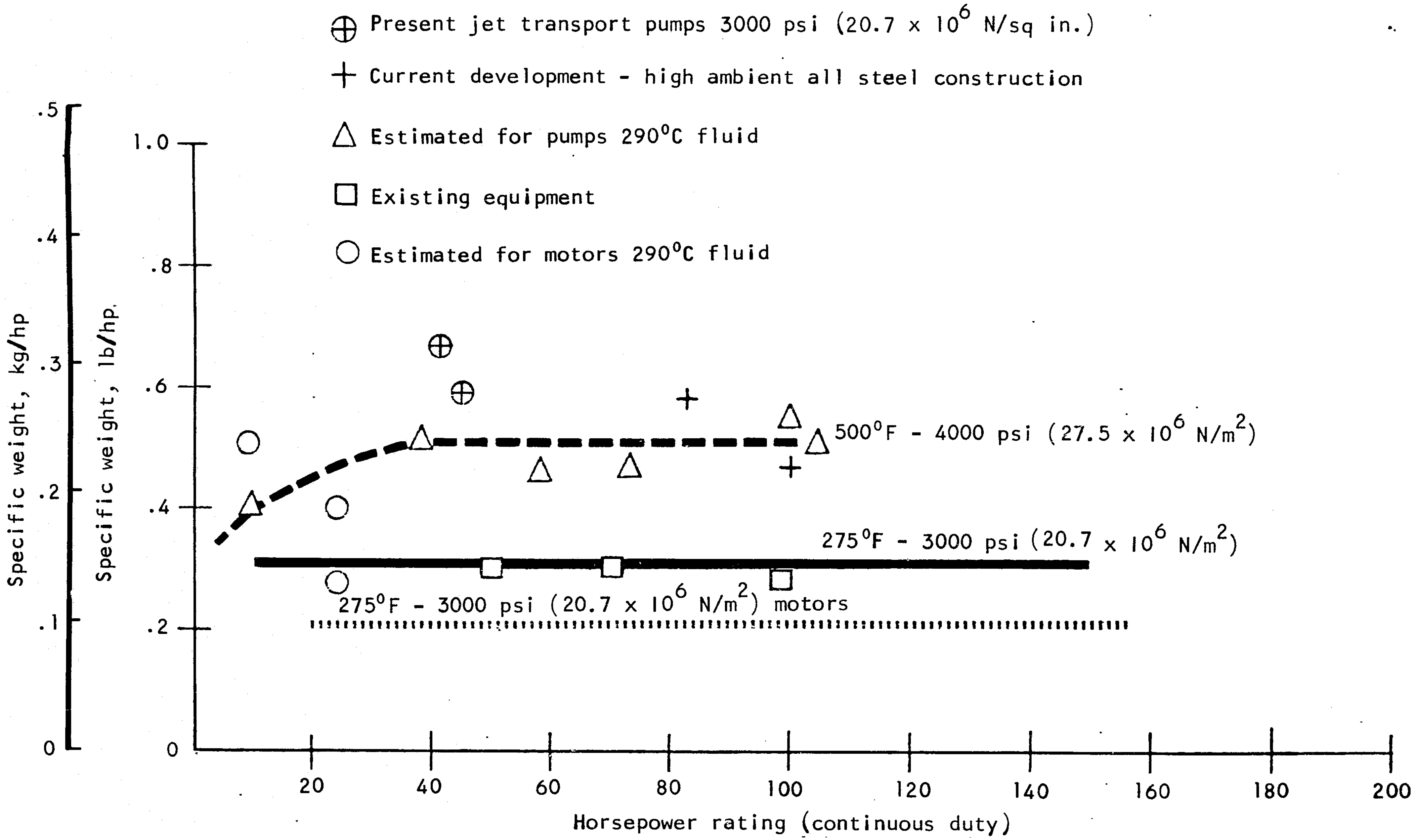

Figure C-10. Pump Specific Weight vs Power (Ref. 48)

Because current jet transports and future supersonic transports have higher power requirements, larger pumps than currently manufactured will be required. Fig. C-11 presents nominal pump performance in the higher flow ranges considering life and power loss as well as weight. Initial tradeoffs can be made from fig. C-11 to determine pump speed and capacity desired vs life, power loss, and weight penalties expected for given flow and system pressure requirements.

One approach (ref. 4) suggested that active cooling of the fluid be provided to minimize the temperature problems associated with the hydraulic system. The largest available heat sink onboard the SST is the fuel supply. A possible way of utilizing this heat sink for hydraulic fluid cooling is the "dual return" system shown in fig. C-12. Cooling systems definitely merit further study and development as one technique to minimize the severity of the SST created environment.

Servovalves.-- Hydraulic power to the output device is controlled by a servovalve upstream of the device. Servovalves can be classified as three-way or four-way, open-center or closed-center, single-stage or multi-stage, and flow-control or pressure-control devices. Internally, they may use rotary valves or spool valves and be pressure compensated or balanced. Any source of power can be adopted to actuate the servo, although electrical or mechanical inputs are the most common.

The servovalve provides precise position control characteristics with high-response rates and high-power amplification capability. They are generally considered reliable devices which can be decreased in volume in spite of the internal complexity.

Many types of servovalves, manufactured by different valve manufacturers, are available for aircraft use. Typical servovalve capacity as a function of supply pressure is presented in fig. C-13 for a series of servovalves. Weight and volume variations for this series of servovalves is presented in fig. C-14.

Hydraulic actuators.-- The hydraulic power output device usually consists of a motor or an actuator. For flight control systems and actuation systems, the hydraulic actuator approach is widely used since the power is directly converted into usable motion. Linear motion is used most widely, but other hybrids exist, such as the digital and the Caravelle "servodyne." In addition, reliability is improved by a redundancy of design where multipistons are located on a single shaft. Each piston has a separate servovalve which operates independently. Failure of one valve reduces the overall performance capability response and rate but enables the system to continue functioning.

The sizing of actuators varies considerably, primarily in accordance with available space, system pressure, and loading. For example, fig. C-15, presents the stroke-to-area ratio of over 300 different flight control actuators used by 11 airframe contractors and over 50 electrohydraulic servo actuators used for various aerospace applications (ref. 5). The variation indicates that no specific constraint is necessary to achieve the desired performance. The variation in actuator design as a function of supply pressure for a given load is presented schematically in fig. C-16.

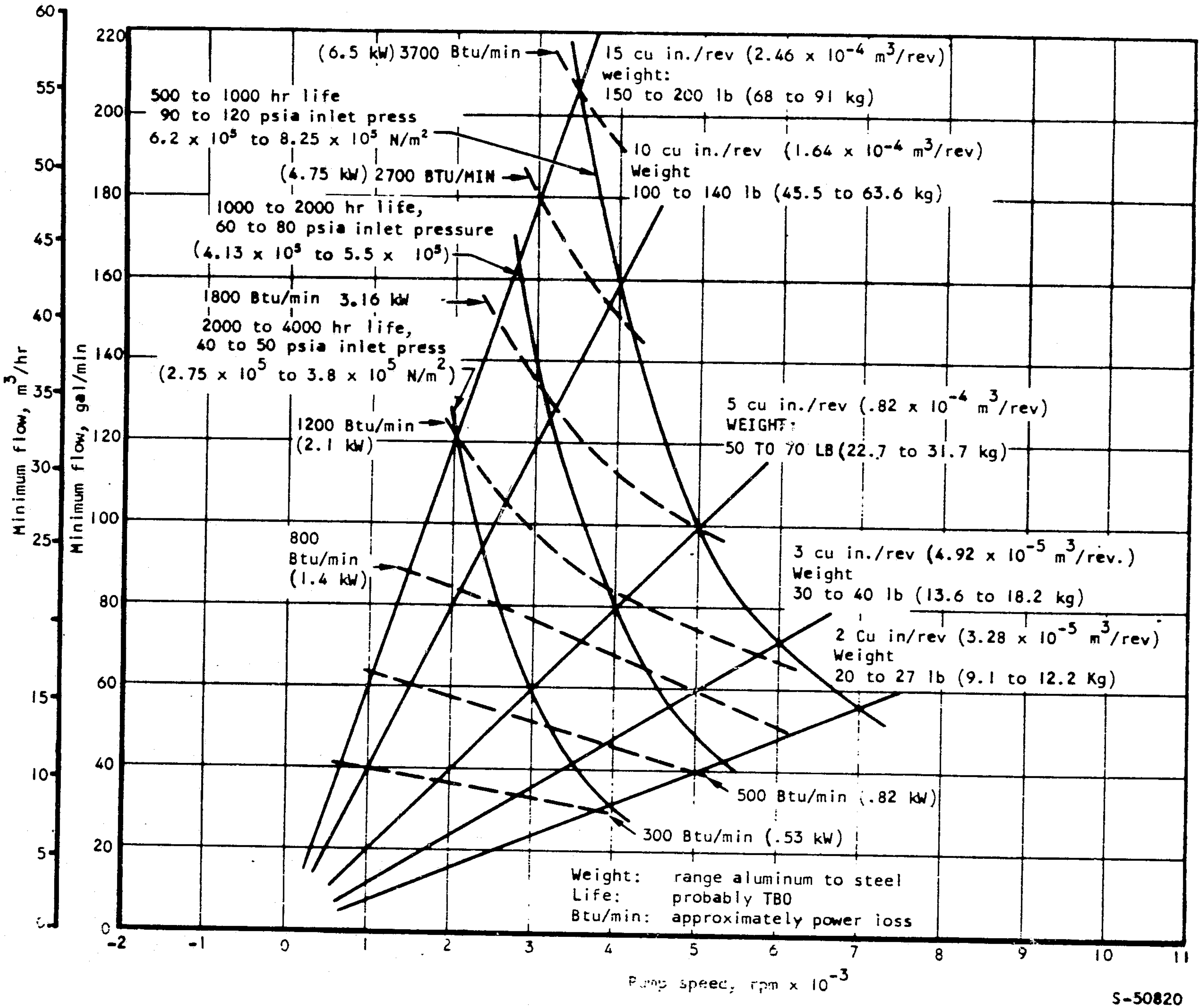

Figure C-11. Nominal Pump Performance Curves

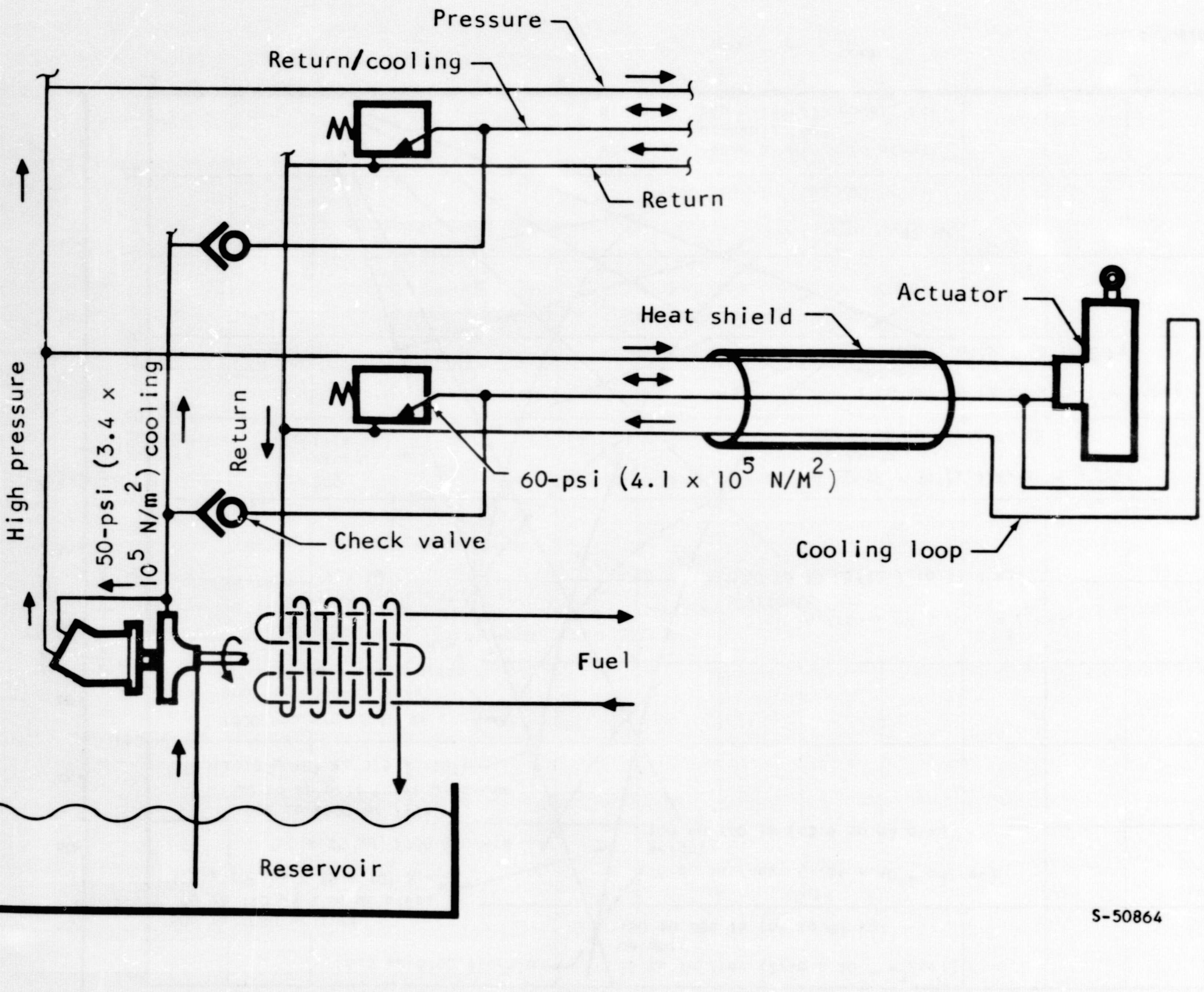

Figure C-12. "Dual Return" Actuator Cooling Concept (Ref. 4)

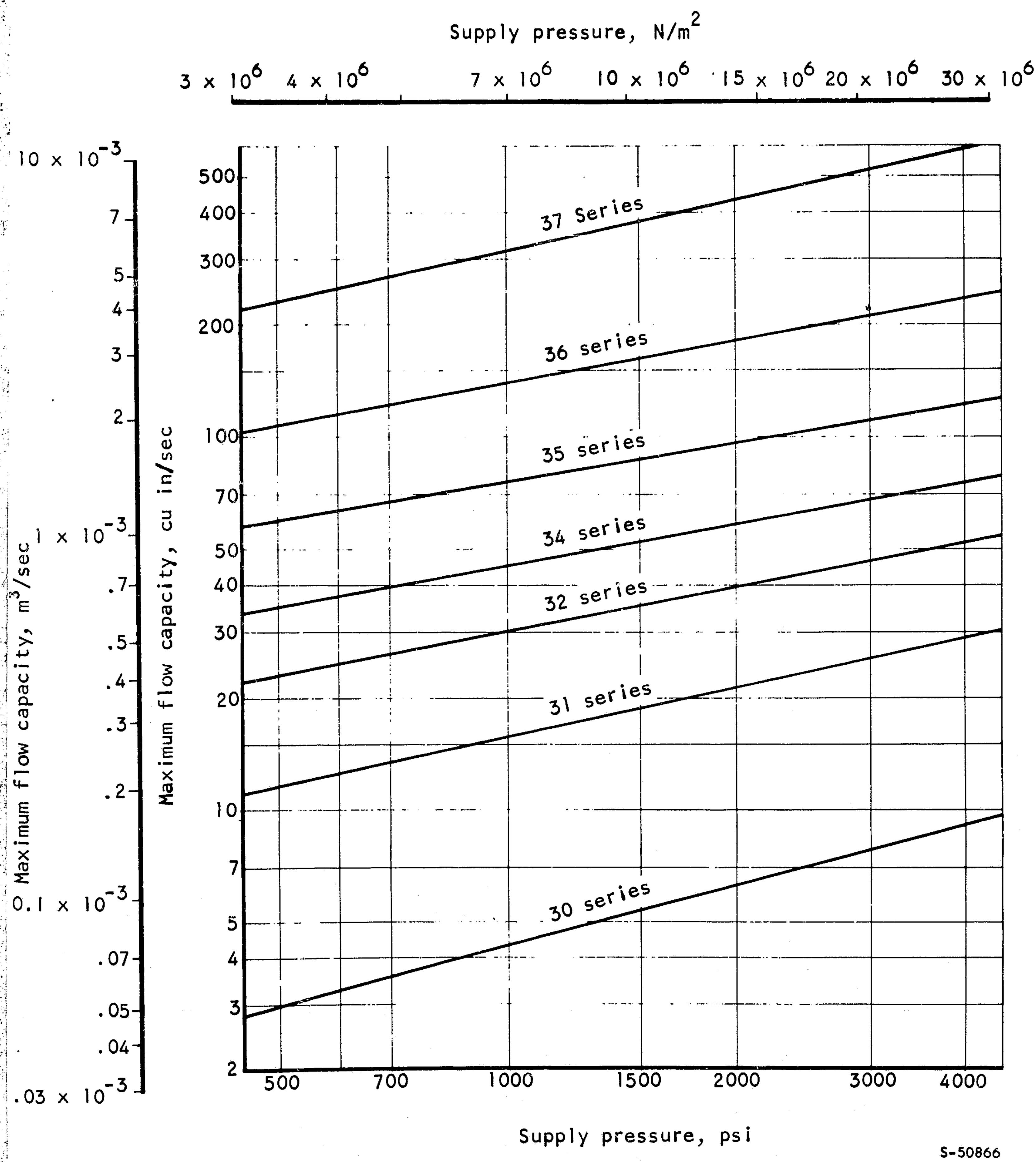

Figure C-13. Flow Capacity of Typical Servovalves (Moog Type 30) (Ref. 5)

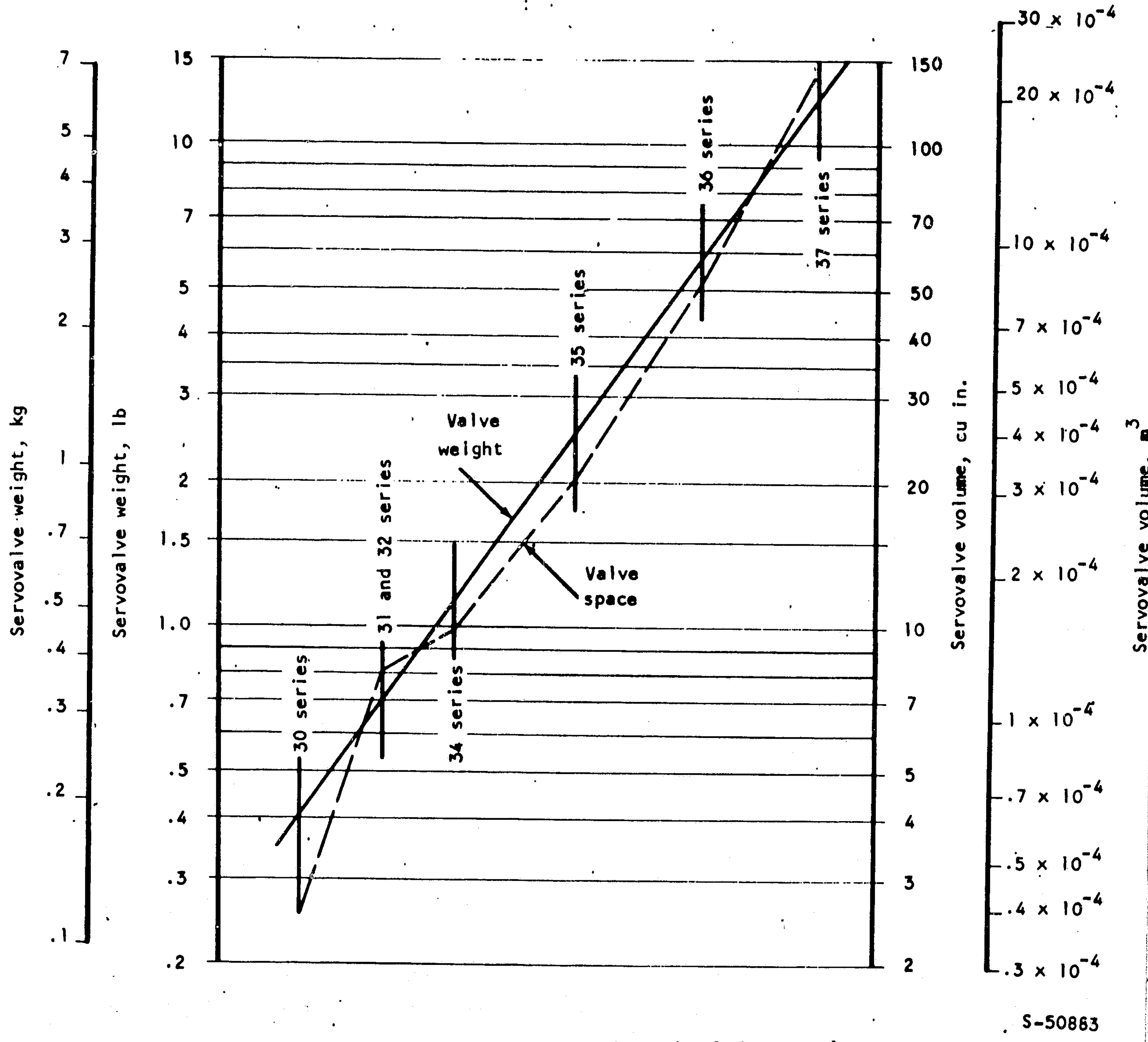

Figure C-14. Size and Weight of Typical Servovalves (Moog Type 30) (Ref. 5)

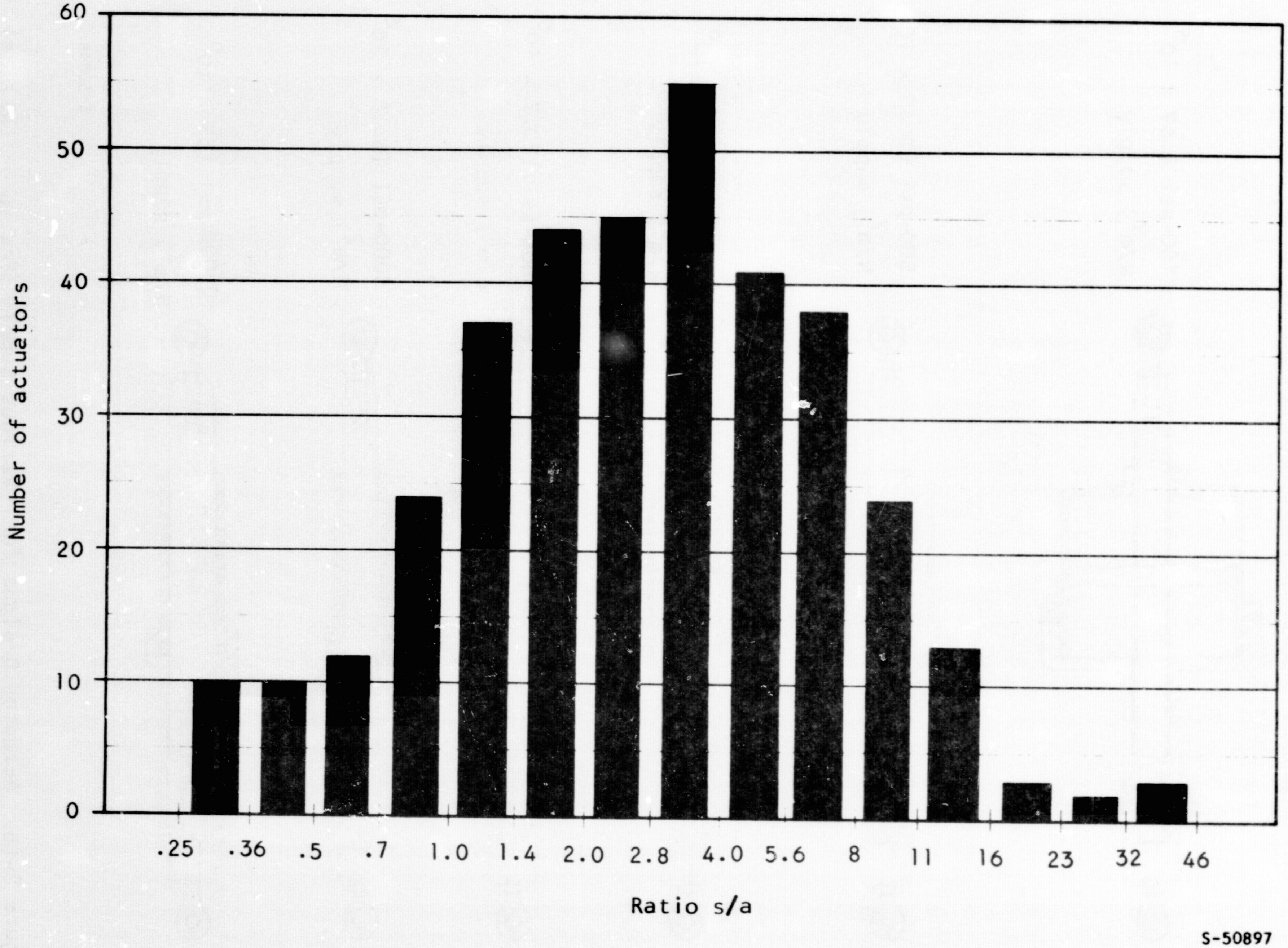

Figure C-15. Stroke to Area Ratios (Ref. 5)

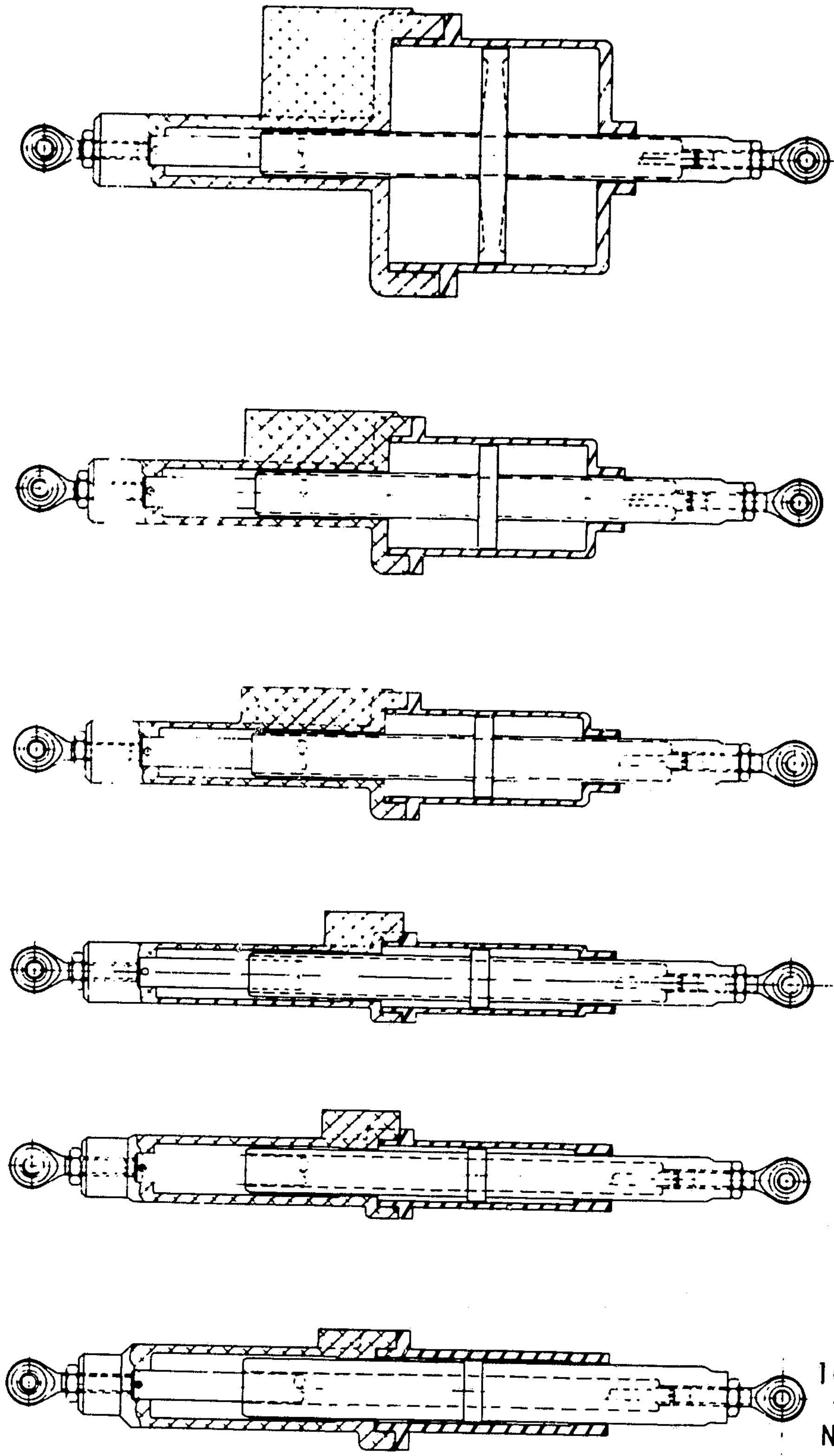

Figure C-16. Actuator Designs for Various Supply Pressures (Ref. 5)

The actuator must be sized to provide the load force at the required rate. Fig. C-17 presents typical operating relationships. Resultant designs yield weight, size, and space tradeoffs as illustrated in figs. C-18, C-19, and C-20.

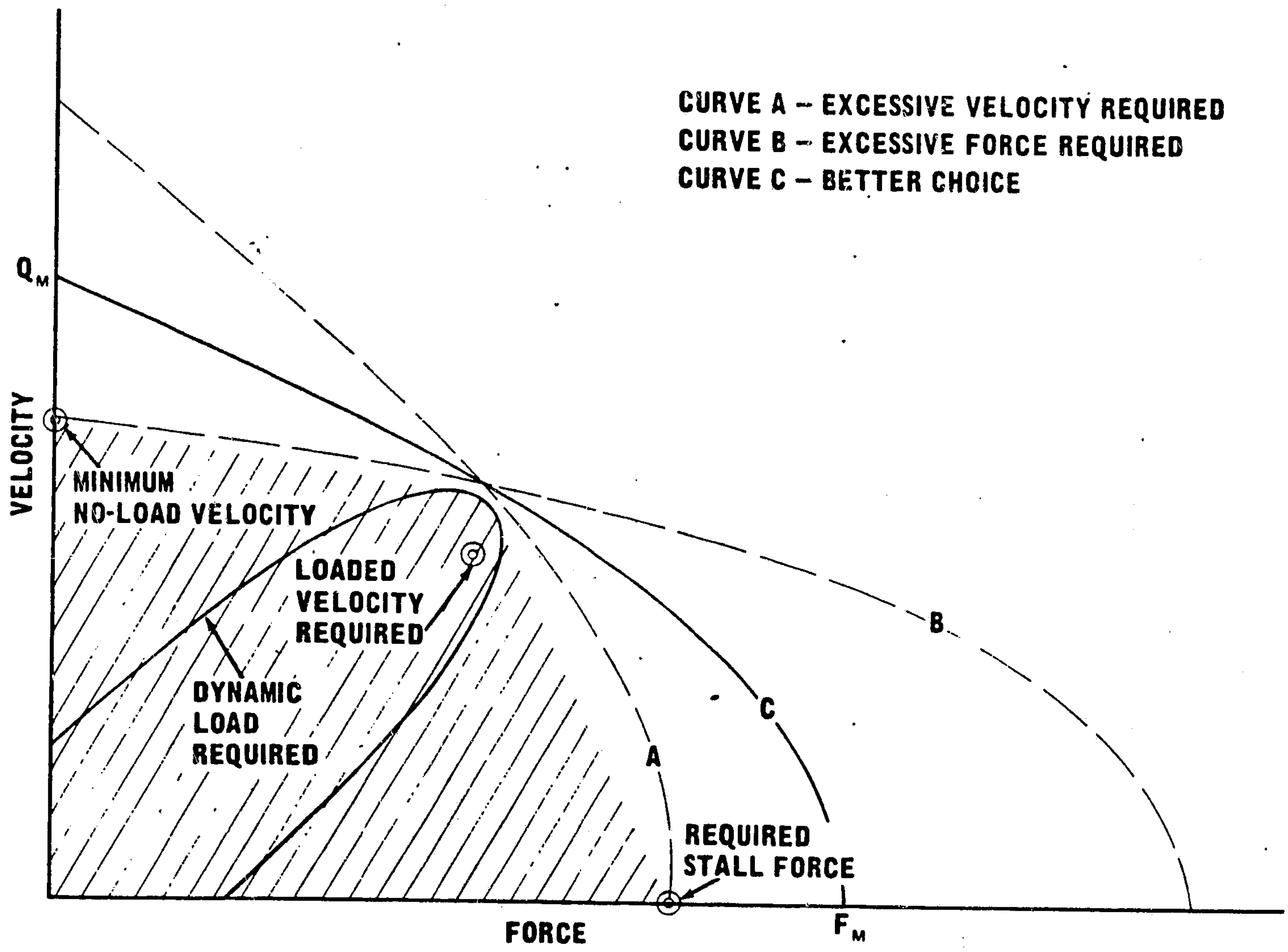

Figure C-17. Typical Load Force/Velocity Requirements (Ref. 5)

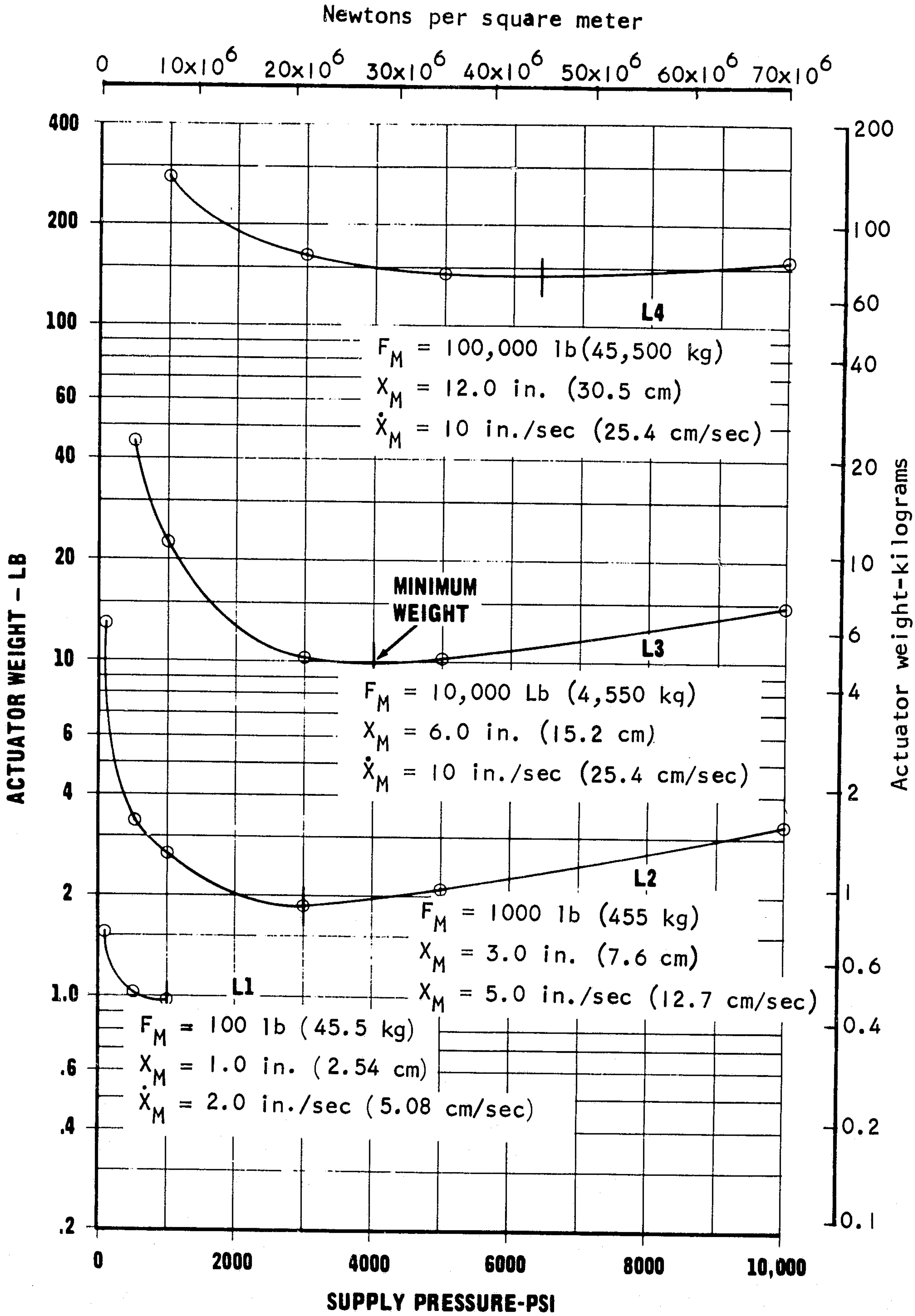

Figure C-18. Actuator Weight Predictions (Ref. 5)

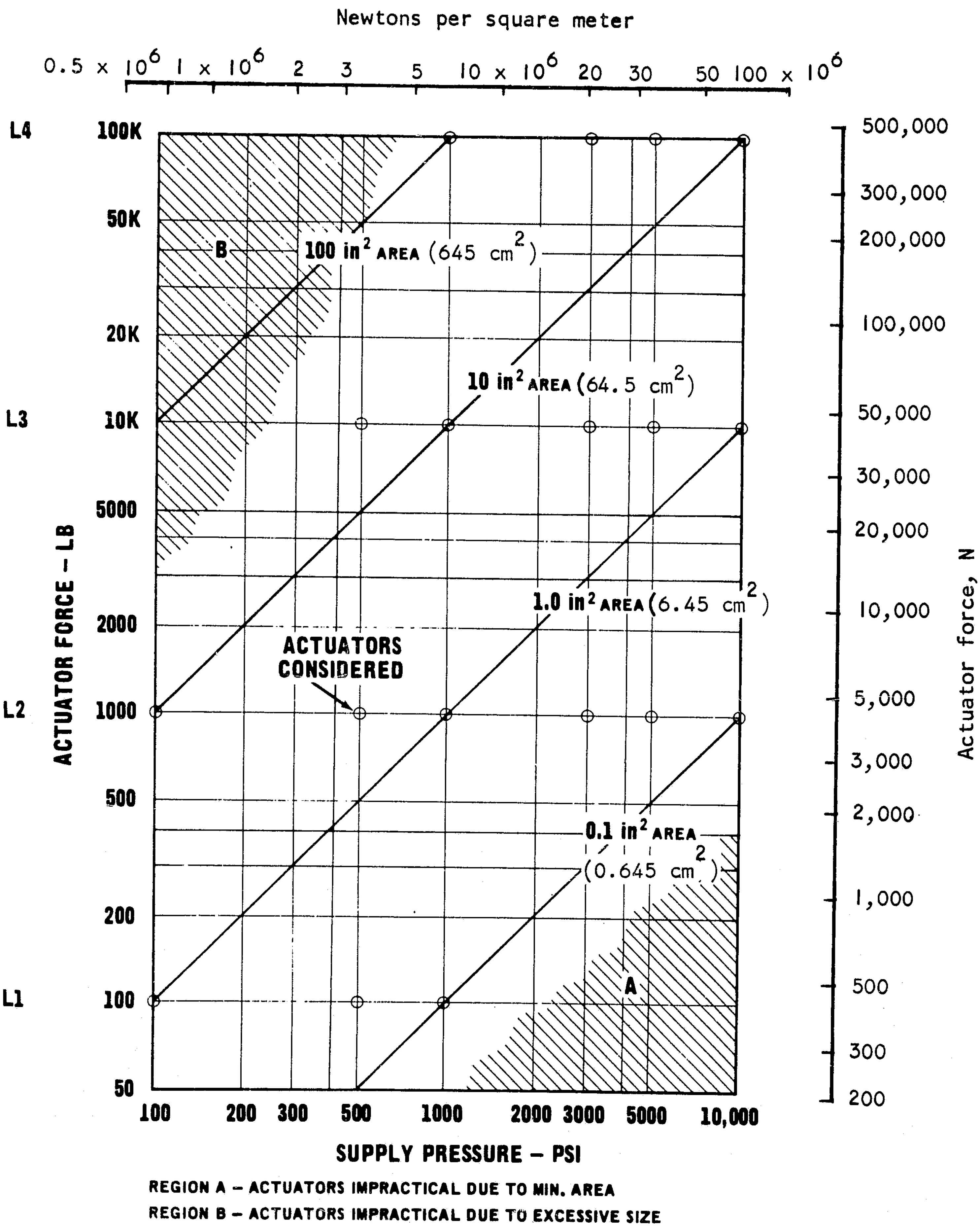

Figure C-19. Actuator Design Sizing Tradeoffs (Ref. 5)

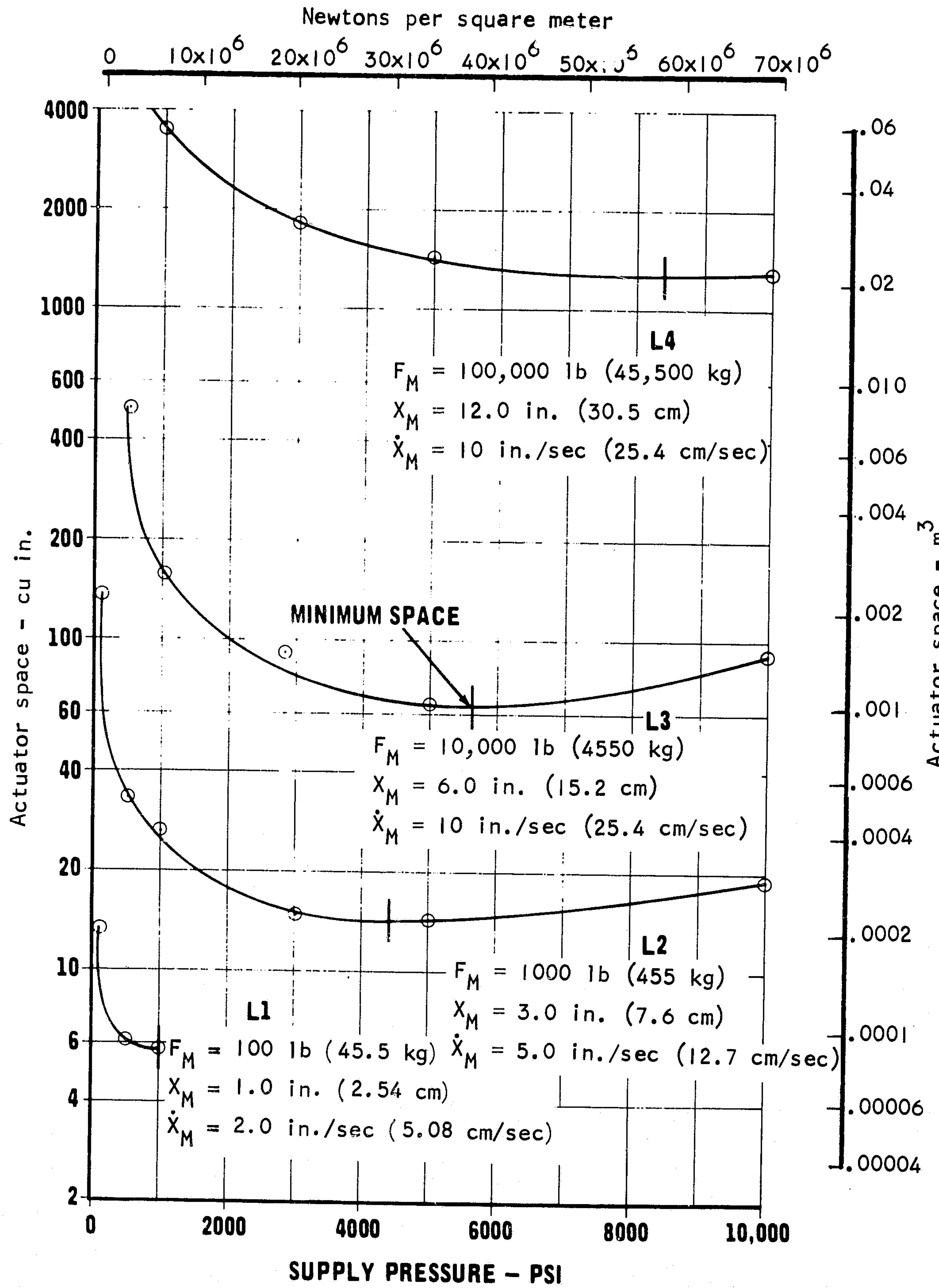

Figure C-20. Actuator Space Requirements (Ref. 5)

APPENDIX D

HEAT TRANSFER METHODS ON EXISTING AIRCRAFT

Component Cooling Methods

Essentially, the cooling methods now being used in aircraft electrical equipment can be grouped into seven categories. Each method is briefly described below.

Self-cooling--By this method, the equipment is cooled by natural convection between the equipment surface and the surrounding fluid. It is always aided to some degree by radiation. Extended surfaces are very often used, and production-type finned heat sinks are available where more convective area is needed.

Power transistors are often cooled by this method. Fig. D-1 shows some arrangements and a serrated surface by which the convective area is further increased.

This method is good when the required heat dissipation rate is in the order of .5 W/cu in. (.031W/cm^3). Conventional motors, generators, and transformers quite often use this method of cooling. Concorde's static inverter also uses this cooling method.

Fig. D-2 shows a self-cooled electric motor. Cast fins are an integral part of the housing. Black paint can be used to promote radiation heat dissipation.

Conduction or heat-shunt cooling--This is a method in which heat is dissipated by connecting the heat source to the heat sink with a good conductor-like metal. Hot spots are avoided by covering the whole component exterior with a metal sheath.

In application of this method, good contacts between parts are not always available. Contact resistances, i.e., conduction through a mechanical joint, can significantly reduce the thermal conductance. In these cases thermal-joint compounds like silicone grease are used to improve the heat transfer.

In cases where high dielectric strength is a requirement, materials like beryllium oxide with safe operation up to 1500-V or more and with thermal conductivity in the same order as that of copper (200 Btu/hr-ft-^{0}F or 1.24 x 10^6 Joules/m-sec-^{0}K) are available.

Fig. D-3 shows several ways that this method has been used in an electronic cooling.

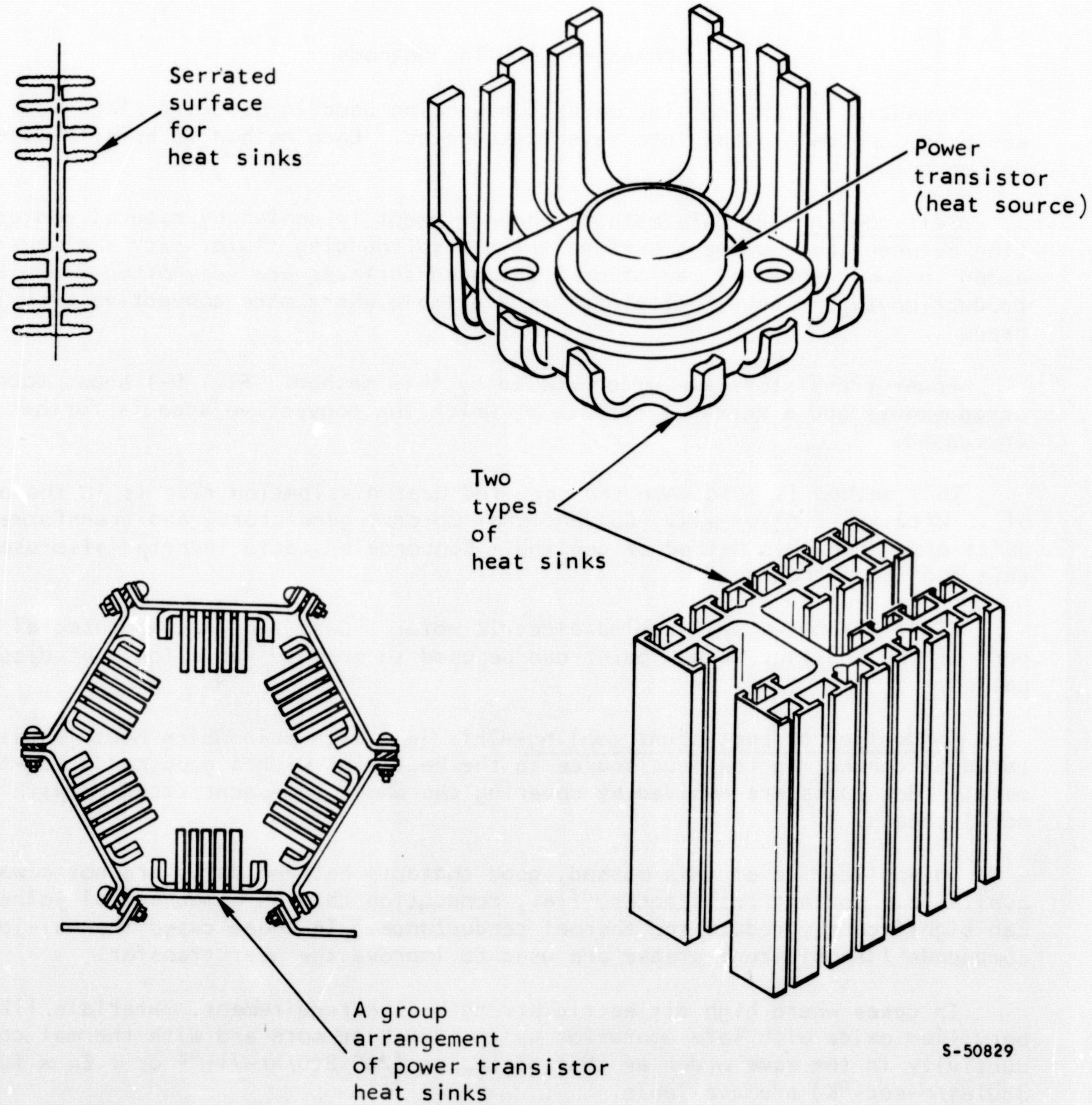

Figure D-1. Power Transistor Heat Sinks for Self-Cooling

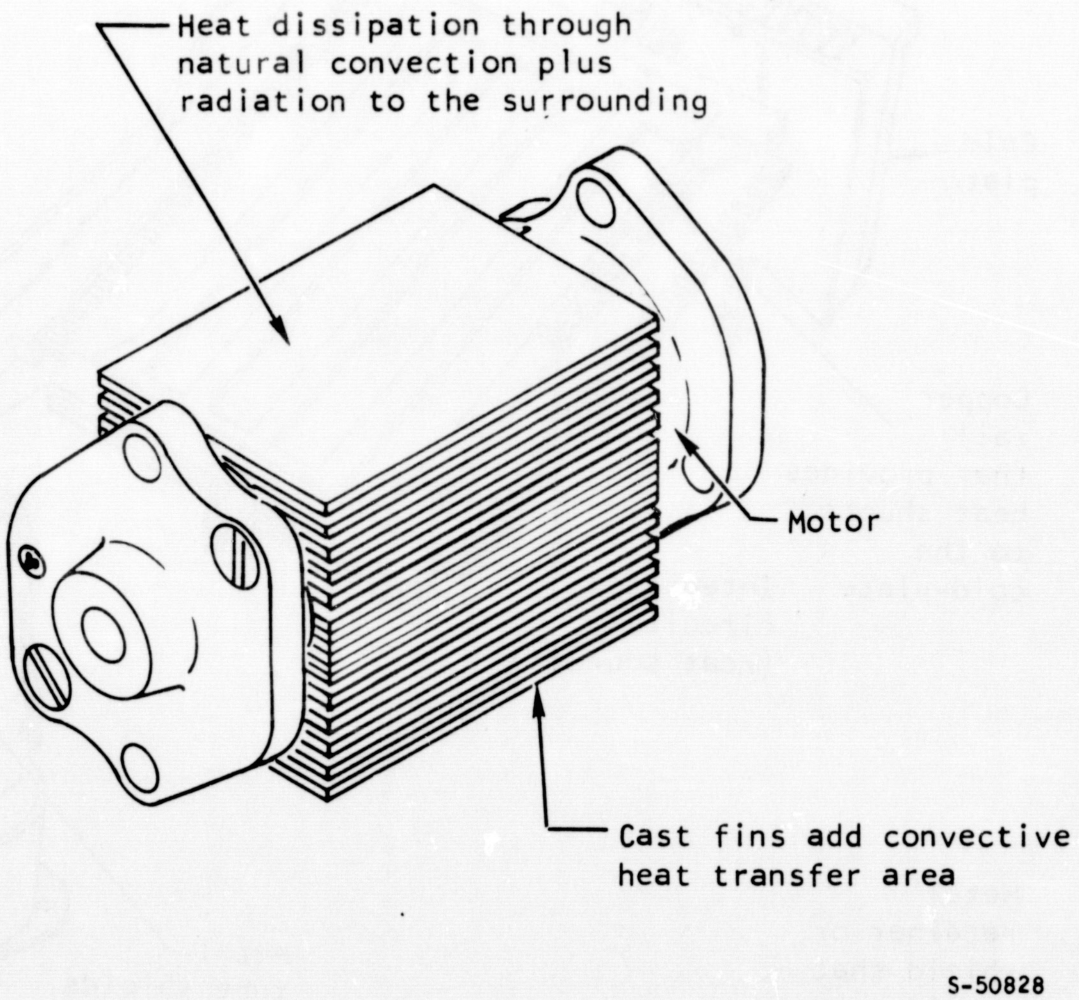

Figure D-2. Flap Actuator Motor

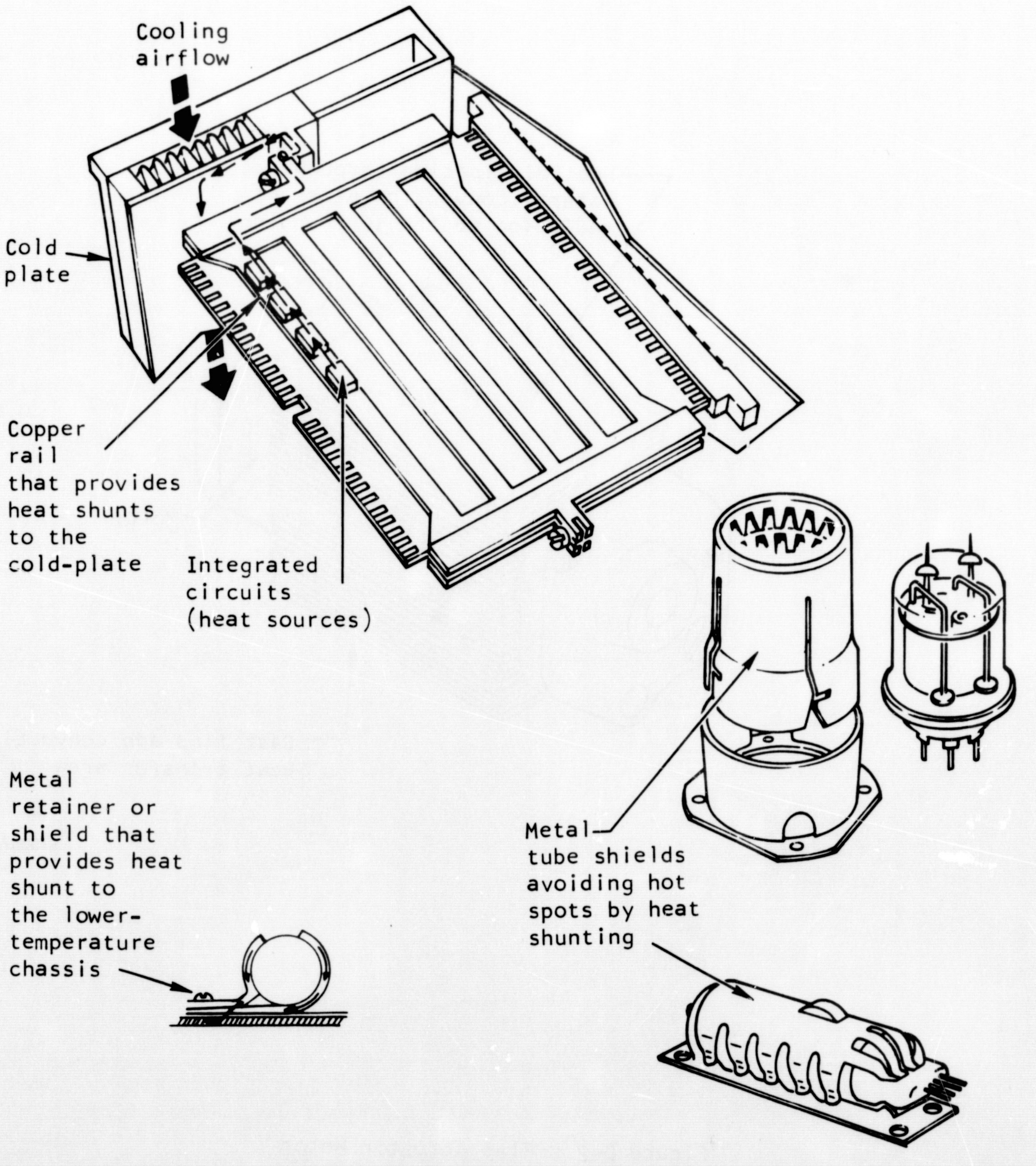

Figure D-3. Conduction (Heat Shunt) Concept and Its Applications

Forced gas cooling--In cases where self-cooling or conduction cooling is not sufficient, the motion of the gas coolant can be mechanically agitated. This widely-used method has an advantage of simplicity because all it requires is a fan or a blower with or without ducting. Conventional motors and generators that have built-in fans are cooled by this method. Concorde's transformer-rectifier unit has an integrated fan and is thermally switched.

To dissipate large amounts of heat, cold plates as shown in fig. D-4 are often provided. The gas from the fan is forced to flow over the equipment and the extended surfaces to dissipate the heat generated.

Another good example of cooling by forced air convection is illustrated in fig. D-5, where the rotation of the rotor is utilized to pump or shear the air for cooling purposes.

High-specific-heat gases, e.g., hydrogen, are preferred to those with lower specific heats because the former require less coolant flow rate and provide a larger convective heat transfer coefficient. Inert gases are preferred to reactive ones for safety and reliability.

Forced liquid cooling--Forced liquid cooling differs from forced gas cooling merely by the replacement of the coolant by a liquid. Liquid cooling normally requires a closed-loop arrangement and thus loses the simplicity of the air system. In most cases, liquid cooling systems require an accumulator to take up liquid expansion in the system, and a heat exchanger to reject the heat from the liquid coolant to the ultimate heat sink such as air or engine fuel.

However, liquid pumps are normally smaller than the fans for air because of incompressibility. Liquids are more readily adaptable when internal coolant passages prove beneficial because the passages can be made very small. In addition, heat transfer rates can be orders of magnitude higher than those in gas systems. When large heat loads are to be dissipated or when high power density (equipment compactness) is a requirement, liquid convective cooling should give a good lightweight system.

Figs. D-6 and D-7 illustrate some techniques of using this method. Concorde's generators are also oil-cooled.

Thermal attenuation (or thermal damping)--This is a technique that is often used to minimize the effects of the varying or oscillating temperatures of the environment, or to shield off the radiation and/or aerodynamic heating. A damper in its simplest form can be just a metal housing. But the optimum weight and the time lag are two items to be considered in design. Melting materials that absorb and reject the latent heat of fusion at a constant temperature give optimum thermal damping for some applications. Another application is to reduce errors in gas temperature measuring.

A damper that provides the constant temperature environment required for temperature-sensitive equipment like gyroscope in inertial guidance systems is shown in fig. D-8.

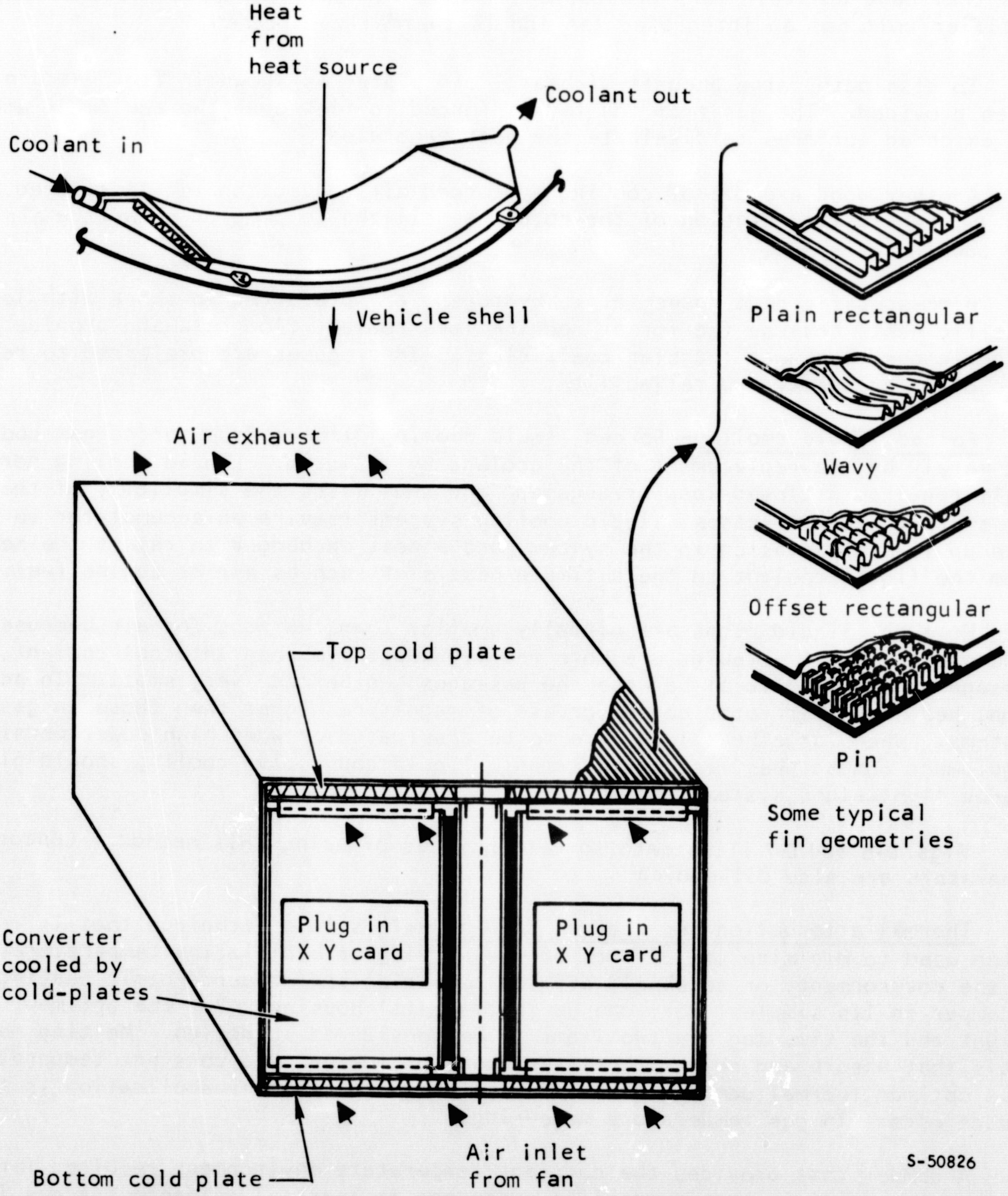

Figure D-4. Forced Gas Cooling and Cold-Plate Concept

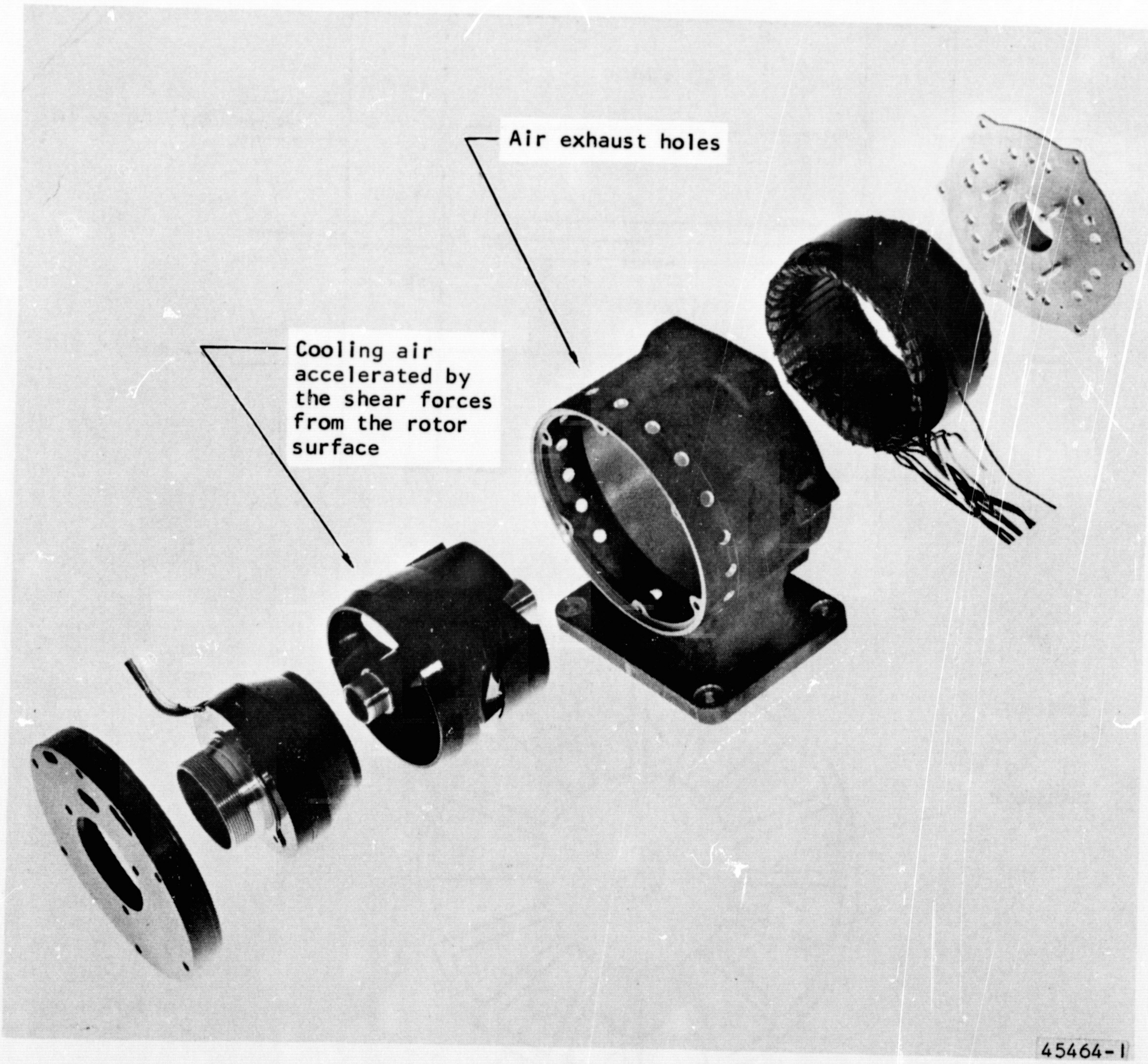

Figure D-5. Forced Convective Cooling by Air Accelerated by Shear Forces (Ram Air Turbine)

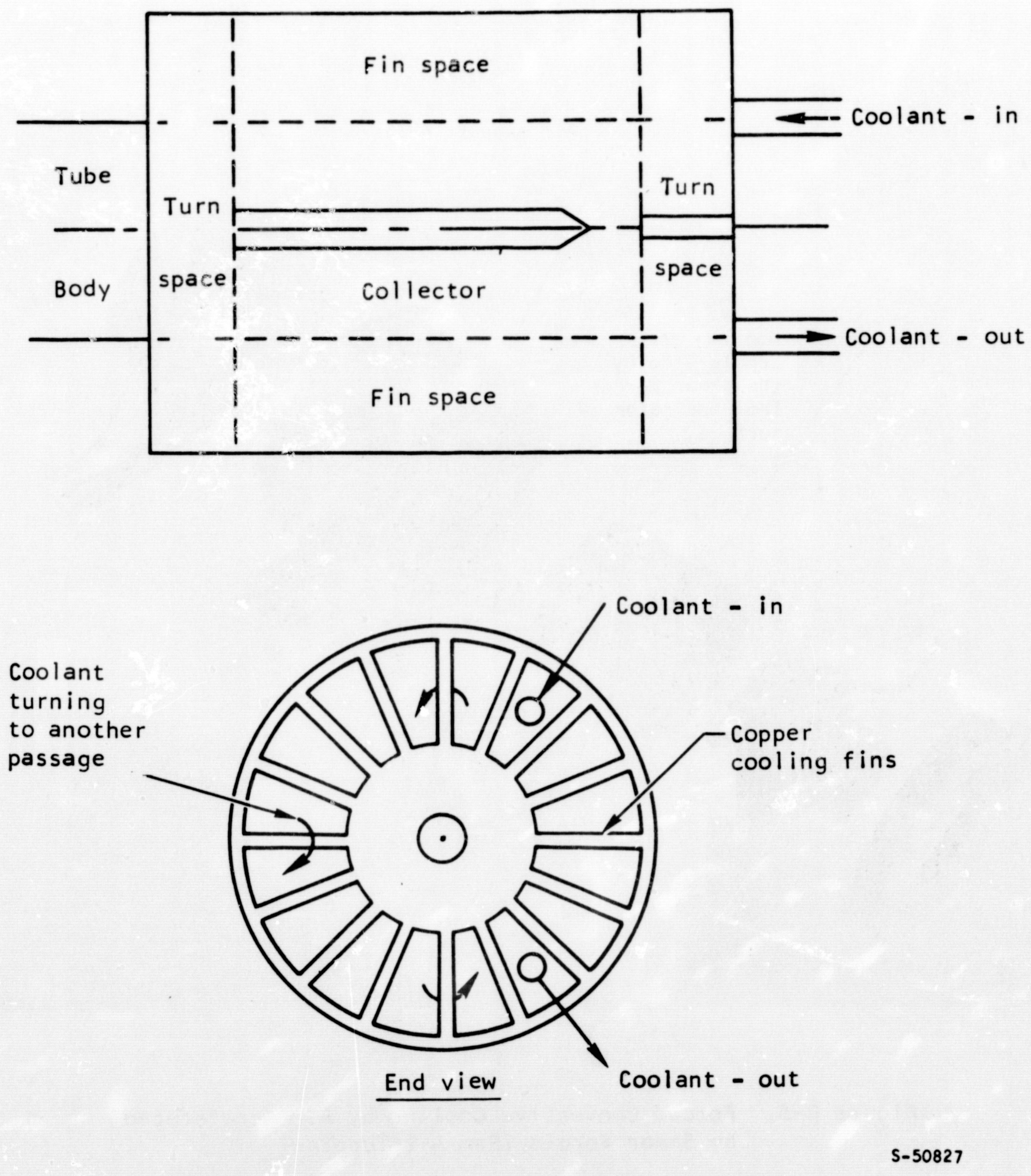

Figure D-6. Forced Liquid Cooling for Traveling Wave Tube Collector

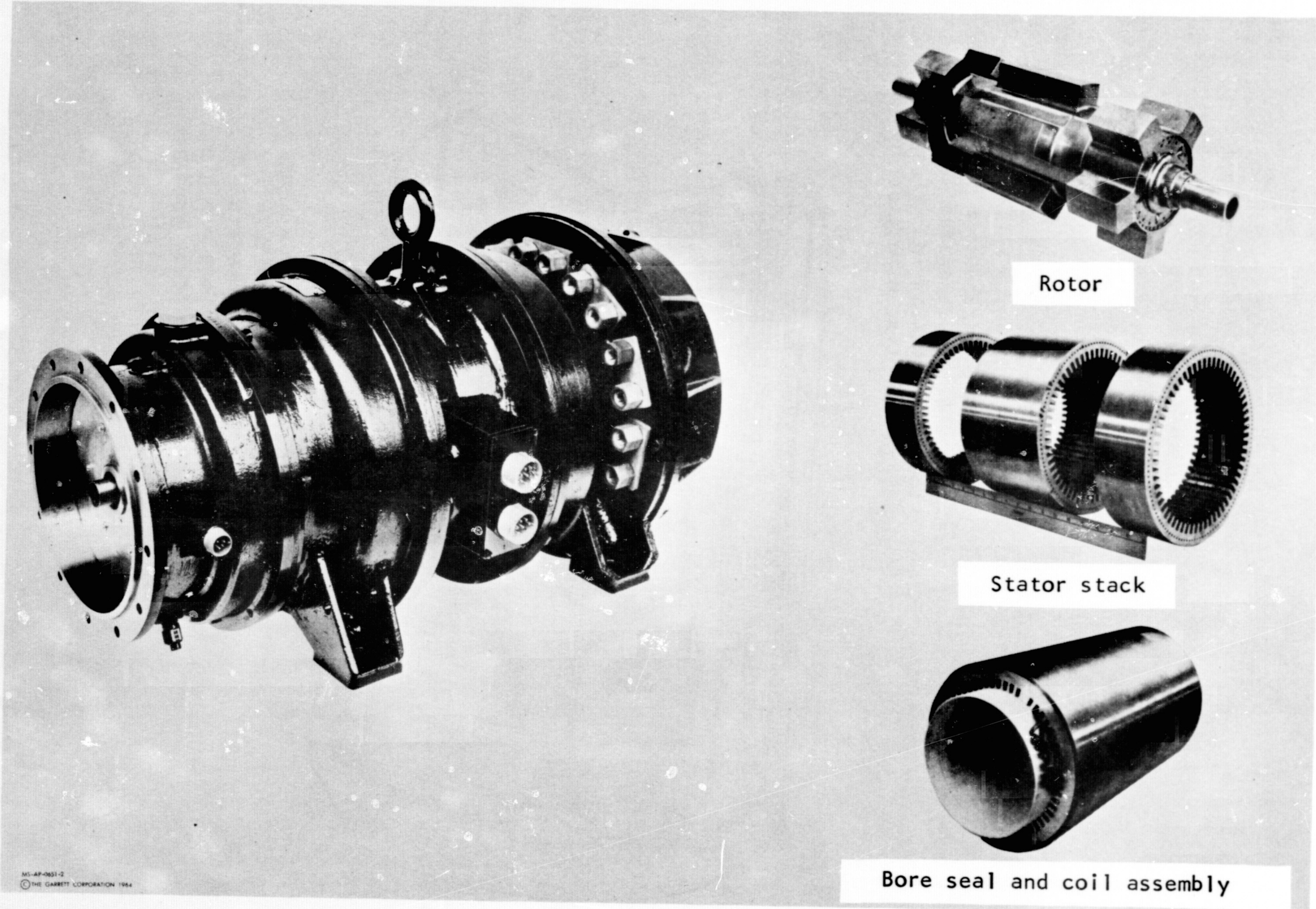

Figure D-7. Forced Liquid Cooling for a Homopolar Generator (by flowing oil through axial holes both in rotor and stator)

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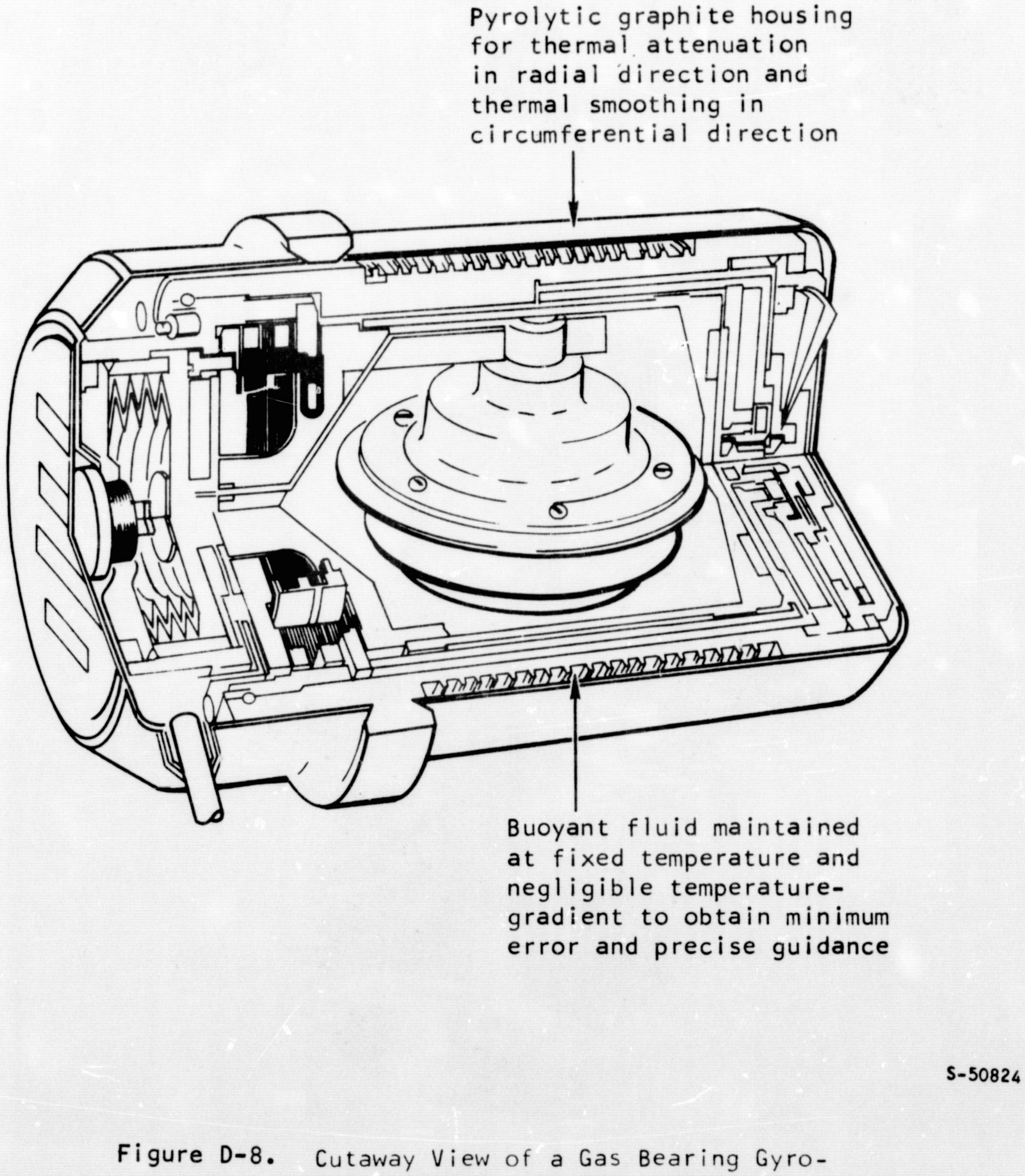


Figure D-8. Cutaway View of a Gas Bearing Gyroscope Showing the Attenuation of Environmental Thermal Perturbation

Thermoelectric coolers and heat pumps--This method or its equivalent is necessary whenever the temperature of the surrounding is higher than that of the allowable component temperature, and no nearby heat sink at lower temperature is readily available. Under such circumstances useful energy is a required input. The application of thermoelectric devices is particularly suited to low heat loads like spot cooling of individual electronic components or infrared detectors. Fig. D-9 shows two units designed for 100-W range.

High-temperature heat exchangers--In high-speed flight or cruise-dash-cruise flight, ram air is often too hot to provide adequate cooling. Two approaches can be taken to solve this problem. One is the design and manufacture of high temperature heat exchangers (for air cycle refrigeration systems). Heat exchangers with metal temperature higher than 1000°C have been developed. Another approach is to incorporate nozzles in the heat exchanger to spray water or other liquids to cool the ram air down before it is used for other purposes.

Cooling Systems

A brief description of nine different aircraft cooling systems is given below. These systems are selected for general considerations and may be impractical in some applications.

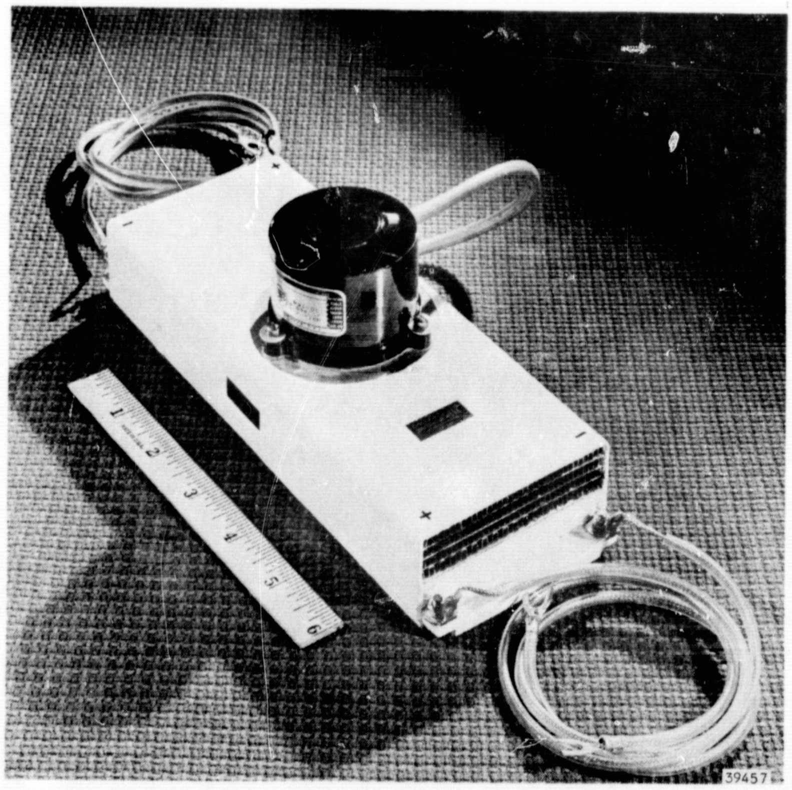
Simple ram air system--In this system, the gas or liquid in the transport loop transfers the heat from the thermal load to the ram air via the heat exchanger and the heated ram air is discharged overboard. A schematic diagram is shown in fig. D-10.

This system is simple. It does not require bleed air from the engine and thus does not affect the engine performance.

When the aircraft speed becomes supersonic, a changeover to fuel as ultimate heat sink can be made or, alternatively, liquids like water can be injected into the ram air stream to reduce its temperature before it reaches the heat exchanger. The system may require supplementary cooling systems at low aircraft speeds and ground static conditions when sufficient ram air is not available.

Expanded ram air system--Fig. D-11 is a schematic diagram of this system, which expands the ram air through a turbine and thus reduces the ram air temperature for cooling purposes. The ram air heat exchanger that is required in the simple ram system is replaced by the cooling turbine and thus is suited for supersonic flight conditions.

The same advantages and disadvantages that are associated with the simple ram air system are applicable to this system, namely, it is simple, but at low flight speeds or ground static conditions supplementary cooling is a requirement. Another disadvantage is that the turbine, compressor, and ducting may become prohibitively large because of the low air densities involved.



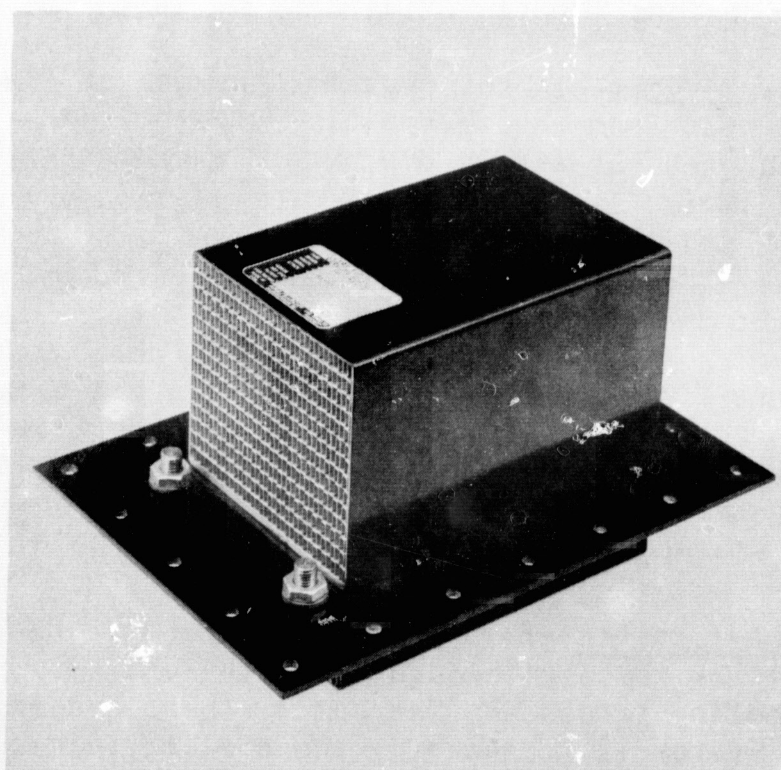


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Figure D-9. Thermoelectric Coolers for 100-watt Cooling Loads

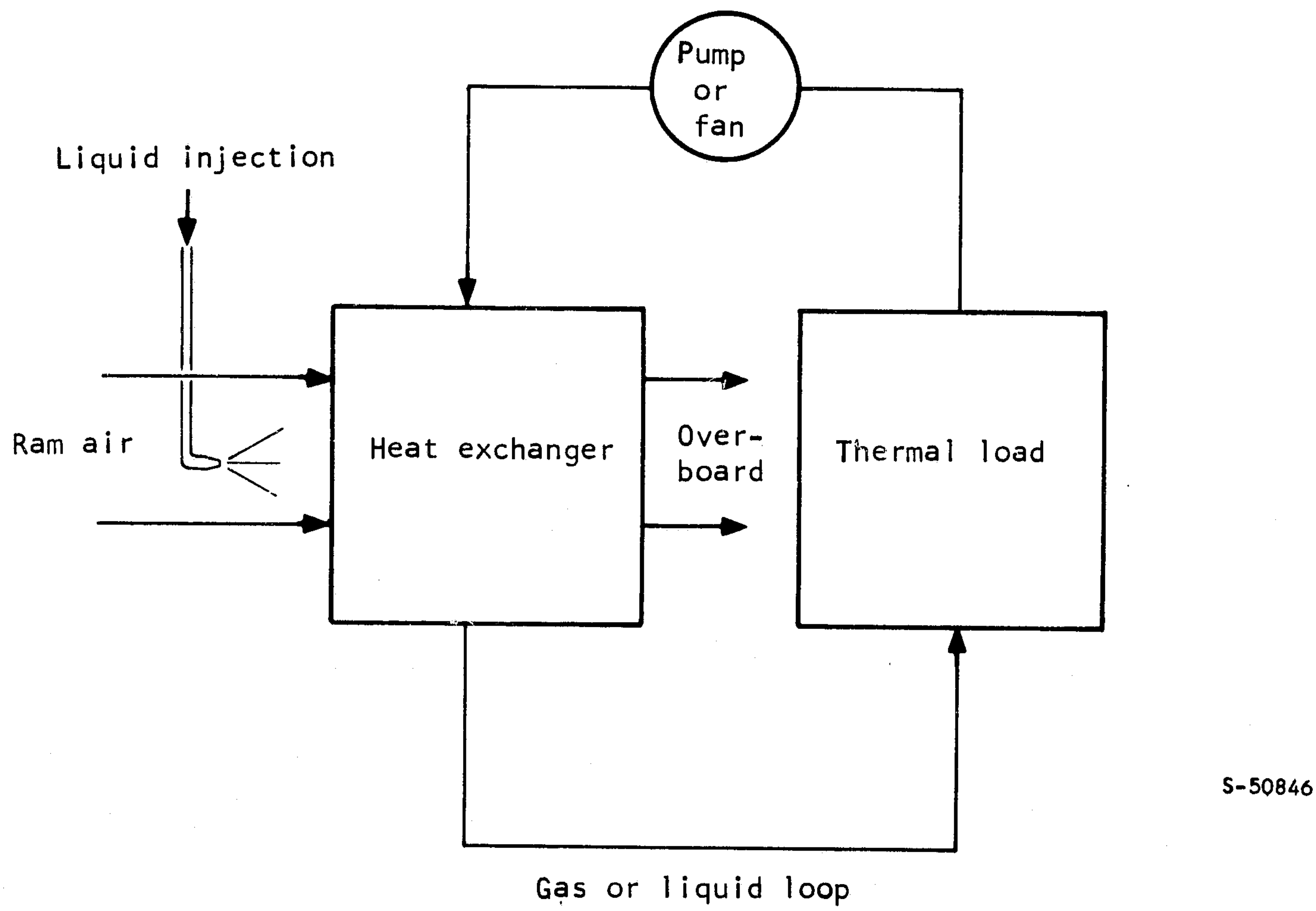

Figure D-10. Simple Ram Air System

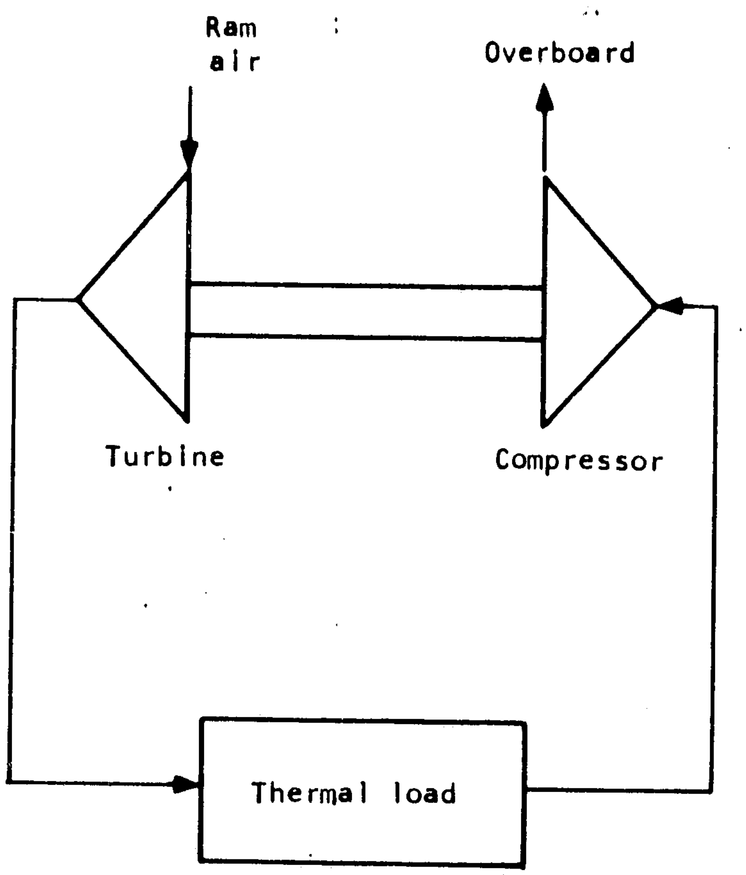

Figure D-11. Expanded Ram Air System

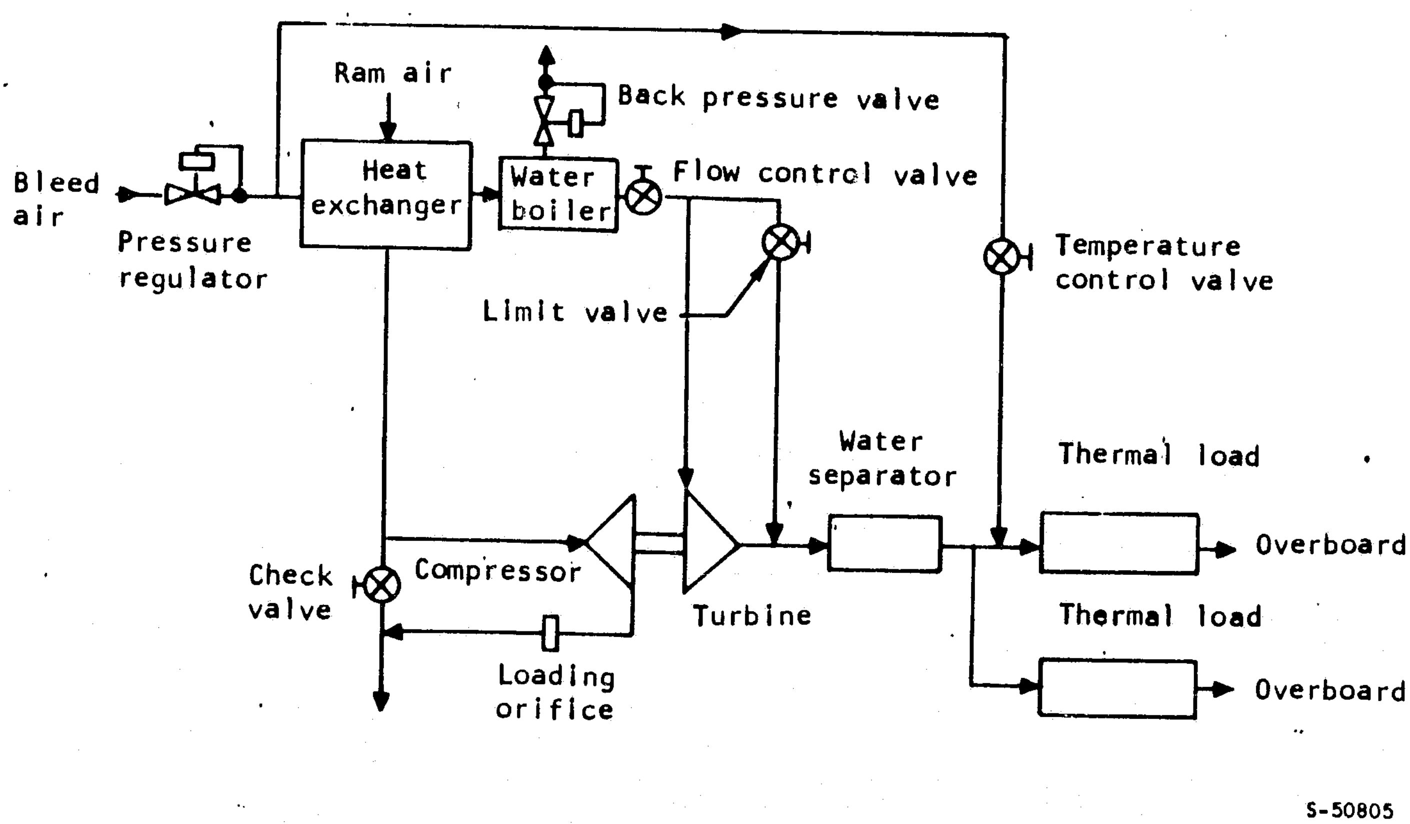

Figure D-12. Simple Bleed Air System

Simple bleed air system--Fig. D-12 shows a schematic of a simple bleed air system. The bleed air from the engine is first cooled by ram air in a heat exchanger before entry into the water boiler. In the boiler, the air is cooled by the latent heat of vaporization of water, thereby further reducing the air temperature. The air then passes through the cooling turbine and water separator and finally reaches the equipment chamber. The turbine drives the compressor which draws ram air through the heat exchanger.

This system is easy to install from the standpoint of controls and ducting. It is simple mechanically and operates with the lowest turbine inlet pressures. A further advantage is that it uses high discharge temperatures and will therefore tend to be free from icing difficulties. This system can provide the cooling on ground.

If the aircraft Mach number is less than 2.0, a simple system is capable of doing the job, and there is no need for the water boiler. If the Mach number is greater than 2.0, a boiler becomes desirable, and if the Mach number is equal to 3.0 or more, the use of fuel as a heat sink is desirable.

If the cooling load variation is in the order of 5:1, the high point being at low altitude and high speed, then a simple system can be used since the bleed pressures for cooling are normally controlled to that ratio to avoid high-temperature bleed air, and the cooling can be dealt with by various types of flow control devices. If the cooling requirements are higher at altitude than at sea level, then pressure regulation may be the best approach for control or a variable area turbine may be adaptable.

Bleed air regenerative systems--Two types of bleed air regenerative systems are shown in fig. D-13. In this system the bleed air from the engine is first cooled by a regenerator, the cold side of which is the same bleed air, but after it has expanded through the turbine and picks up the electric or electronic heat loads. The cooled bleed air can be used to cool the equipment directly as in part (a). A closed gas or liquid coolant loop can be incorporated as in part (b). With the scheme shown in (a), a water separator may become necessary if moisture presents a problem in the application.

This system eliminates the ram air cooling completely, and therefore is not restricted to subsonic flights. It has performance similar to that of the simple bleed air system and also provides cooling on ground.

Bootstrap air cycle system--If the cooling load is relatively constant (±20 percent) over the flight envelope of the aircraft, and if the air being used for cooling is at a constant supply temperature, then the mass flow required will be essentially constant with altitude. This would suggest the use of a bootstrap type of system with a constant gage inlet pressure regulator, a schematic diagram of which is shown in fig. D-14. This type of system, when sized properly, will provide a flow variation of approximately 20 percent under flight conditions.

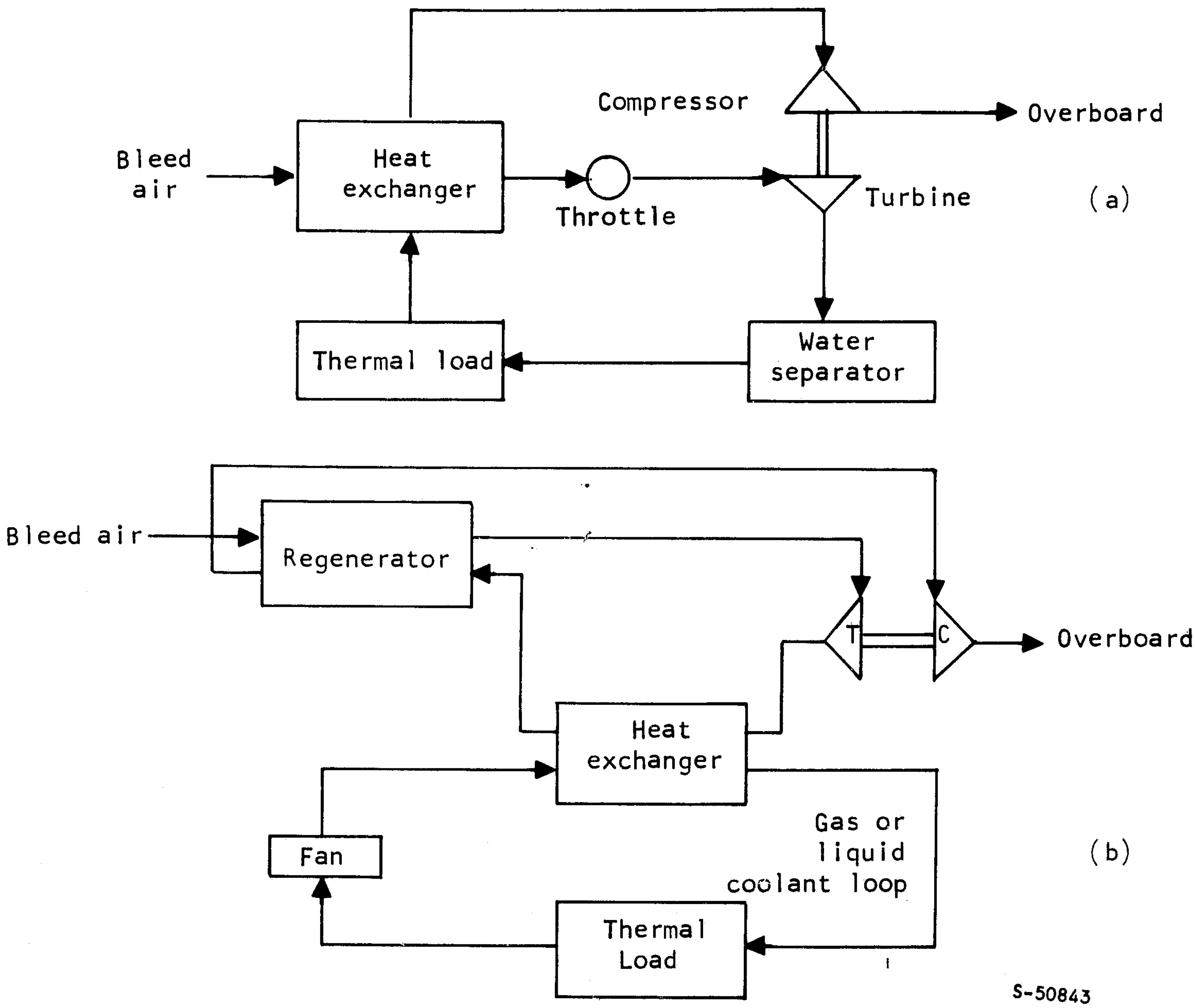

Figure D-13. Bleed Air Regenerative System

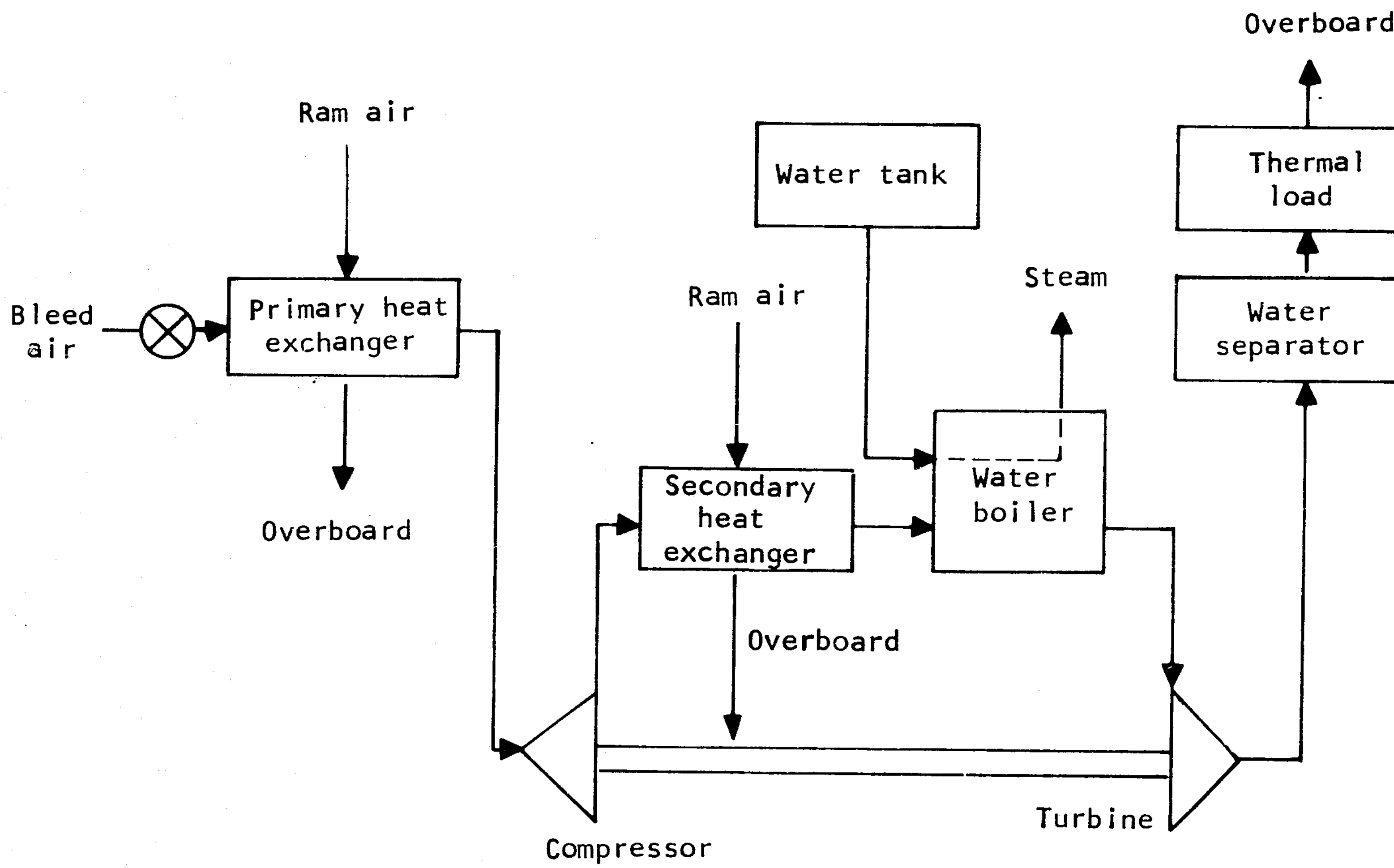

Figure D-14. Bootstrap Air Cycle System

This system is more efficient than the simple bleed air system for a given amount of high pressure bleed air but is more complicated mechanically.

Fans or jet pumps may be put in the ram air stream for low speed conditions or cooling on ground.

In cases where low moisture content in the cooling air is a requirement, one high-efficiency water separator or two in series can be put in. In addition a reheater which increases the air temperature beyond moisture saturation but not over the required component ambient temperature of the electronic equipment can be incorporated.

Fuel cooling systems--Fuel cooling systems utilize transport liquids to transfer the heat from the equipment to the fuel via a heat exchanger are shown in fig. D-15.

Since the same amount of fuel must be carried by the aircraft even with the electrical and electronic equipment generating zero heat, fuel is used as a heat sink whenever it is feasible or safe to do so. A liquid-to-liquid heat exchanger is in general smaller than the air cycle heat exchanger, and the system has all the advantages of forced liquid convection and even boiling/condensing processes. The fuel cooling system may very likely be a lightweight system.

Fuel temperature usually rises during letdown at the end of each mission. Special thermal conditioning to the full storage environment or special cooling consideration to specific items of equipment may be required during this period.

Compact heat exchangers and electronic cooling packages are shown in fig. D-16. These packages include the heat exchangers, fans or pumps, and necessary ducting.

Expendable systems--The expendable systems make use of the large heat transfer rate of boiling process and the large heat of vaporization of liquids to cool the equipment down to a desired constant temperature by presetting the expendable pressure.

Two such systems are shown schematically in fig. D-17. Pressure regulation of the expendables is normally a requirement. Pressurization of the transport loop (right fig. of fig. D-17) may be necessary when leakage is a possibility or the fan gets too big. The biggest drawback of this system is that the required amount of expendable may be prohibitively high for long supersonic flights. The expendable systems are most suited, therefore, when used in conjunction with other systems. It is particularly attractive when cryogenic temperatures are required, but the refrigeration plant is too much weight penalty to be practical.

Fig. D-18 shows two expendable system arrangements that can be applied to electronic cooling. Reservoirs can be used to extend the service time period as shown in part (a).

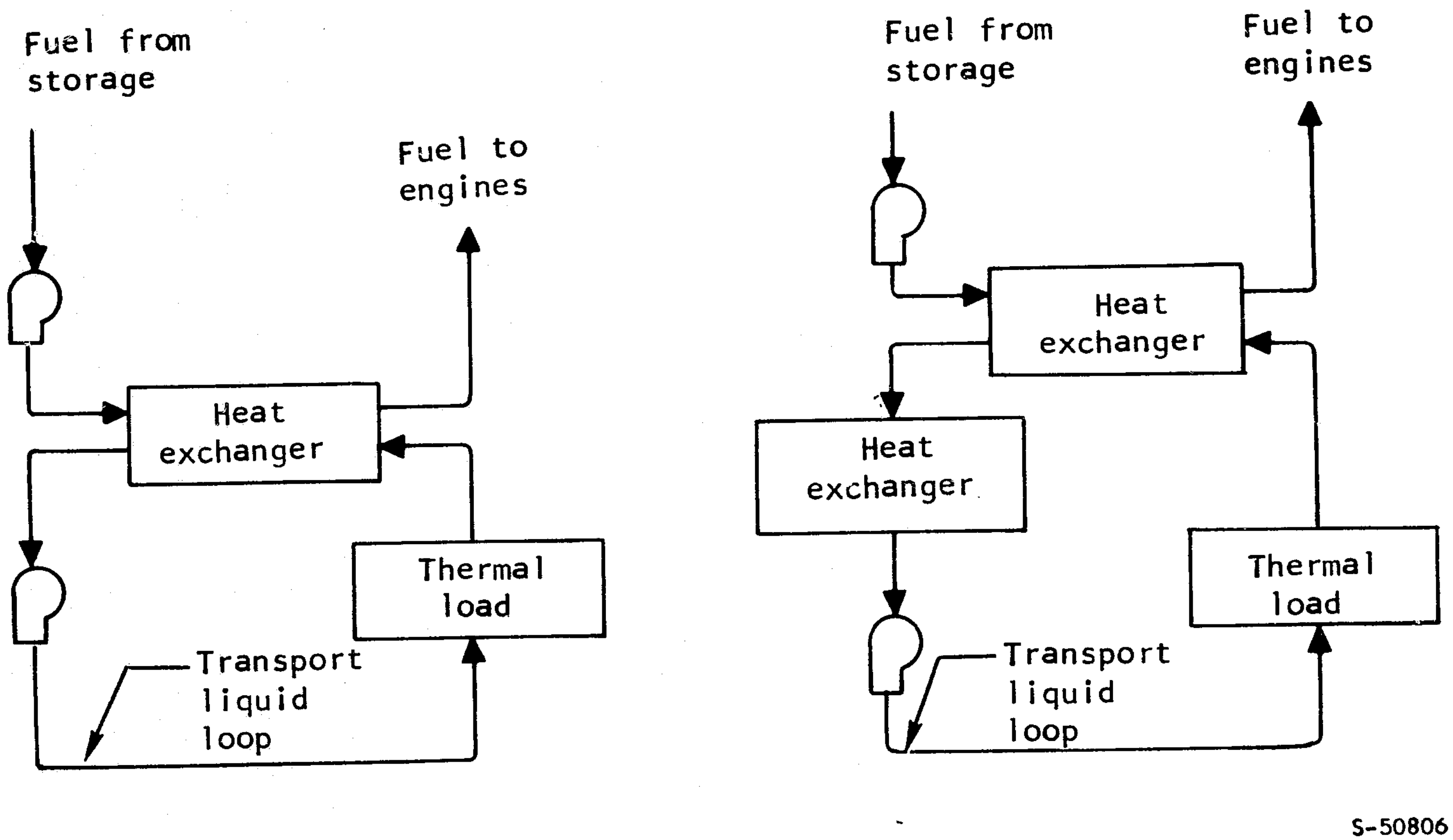

Figure D-15. Fuel Cooling System

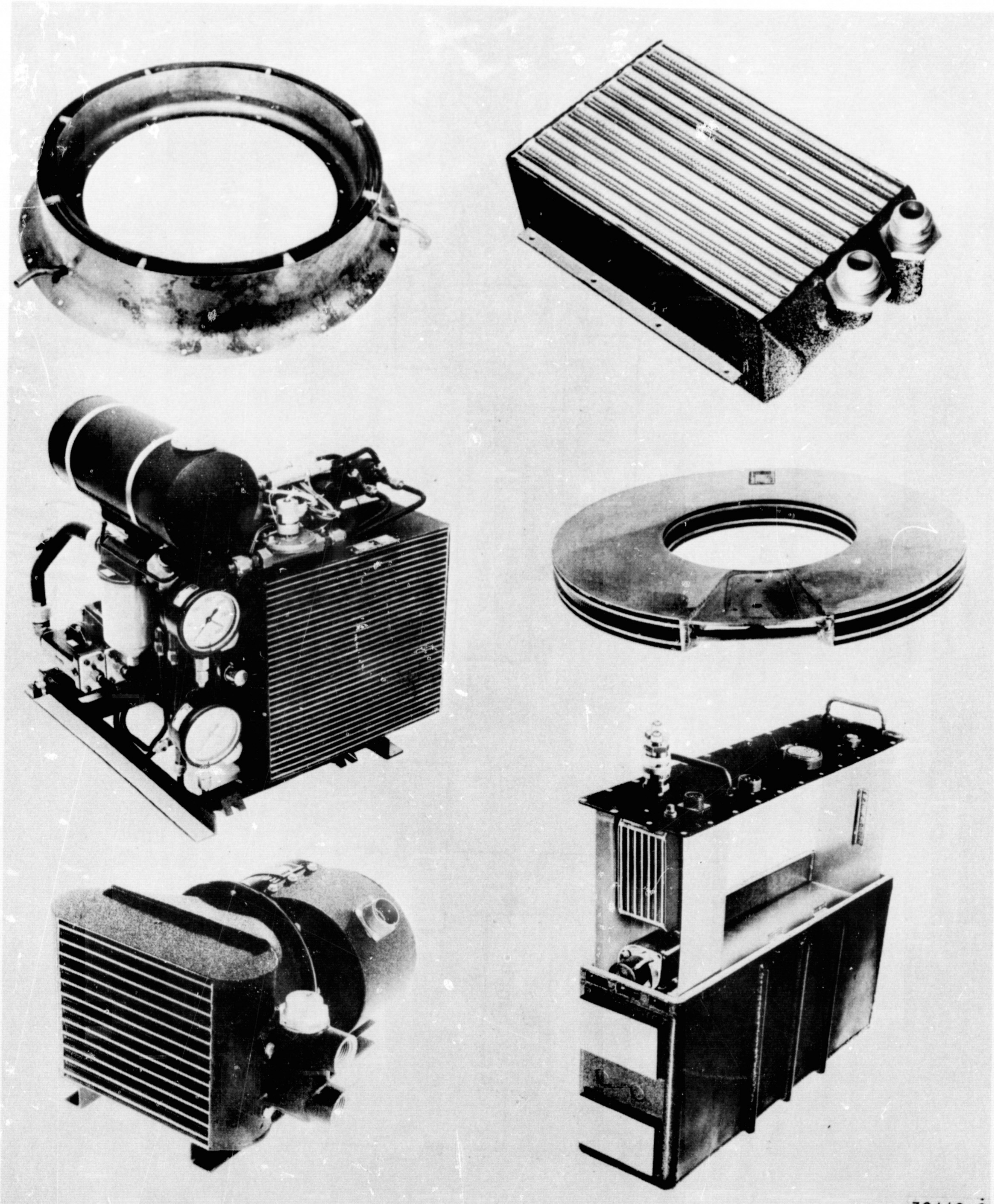

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Figure D-16. Cooling Packages Consisting of Fans, Heat Exchangers for Gas Loops and Pumps and Plumbing for Liquid Loops

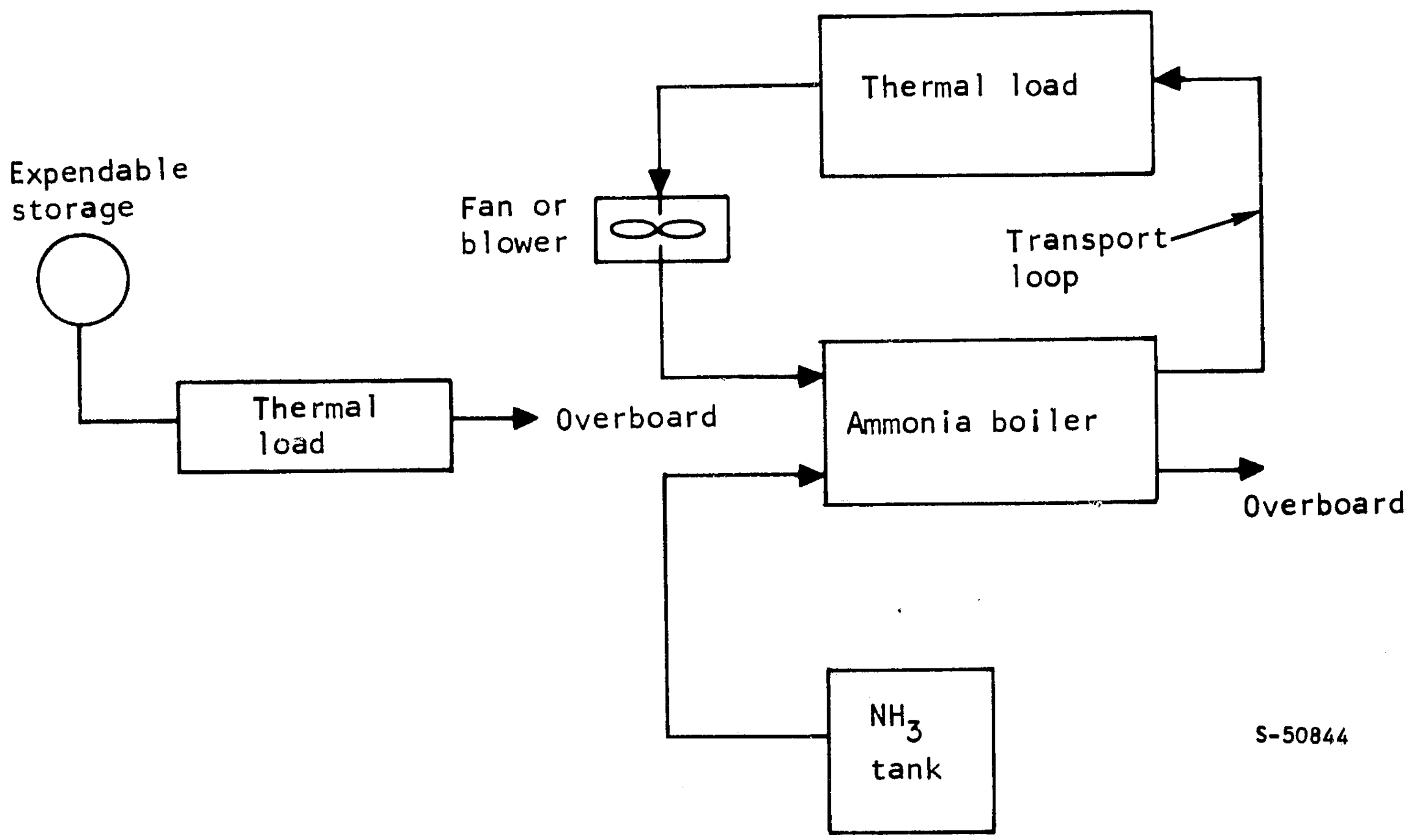

Figure D-17. Expendable Systems

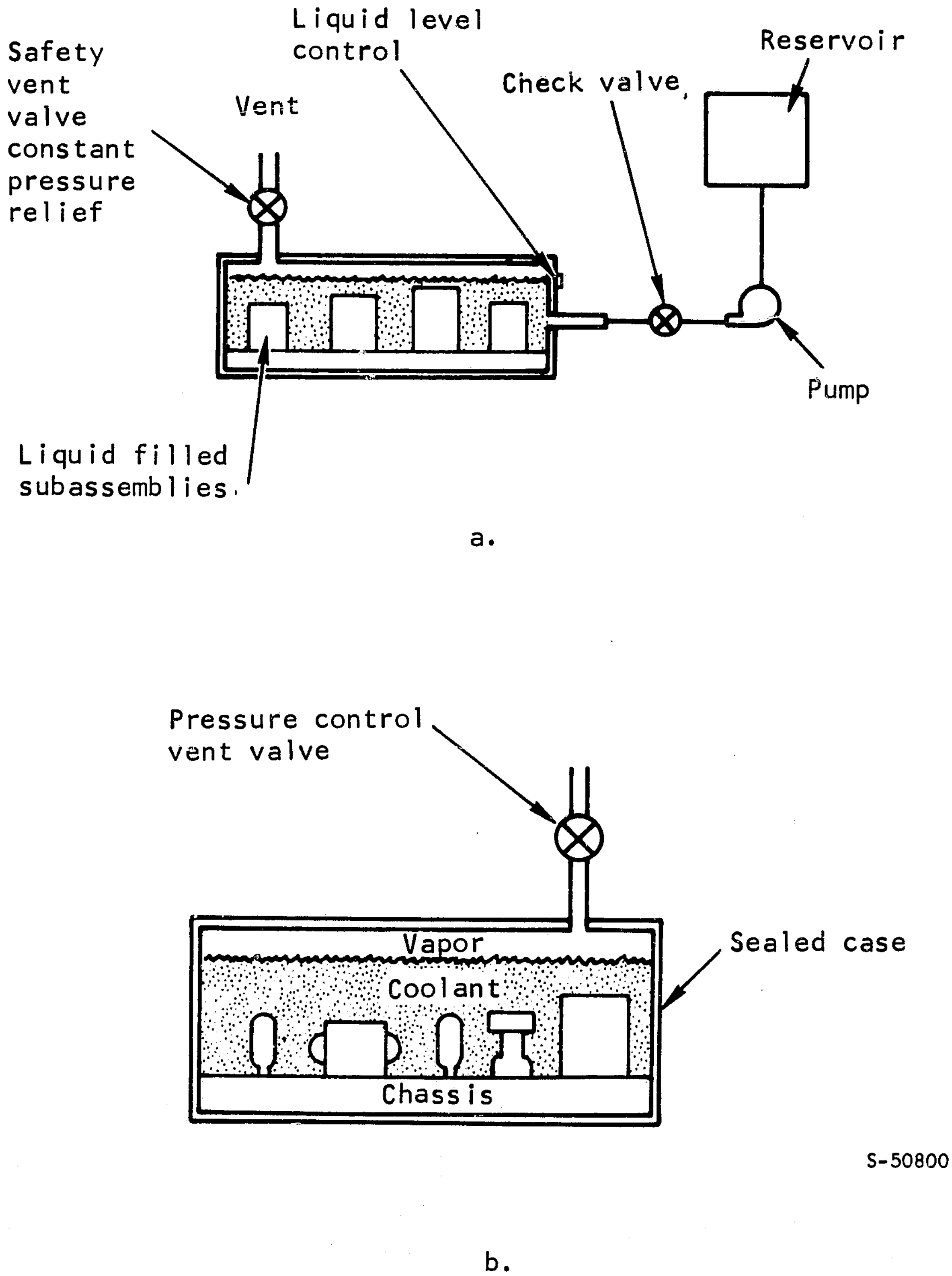

Figure D-18. Direct Expendable Vaporization Cooling System

Vapor cycle system--In essence, the vapor cycle system uses the principle of refrigeration to provide the required cooling. One possible configuration is depicted in fig. D-19 where a closed loop for the refrigerant (vapor loop). another closed loop for the transport fluid (liquid or air loop), and an open loop for the ultimate heat sink are shown. In this particular arrangement the ultimate heat sink is shown as a combination of ram air and expendables.

This system is especially good when bleed air or low-temperature ram air is not accessible for cooling purposes, but shaft or electrical power is available. Small cooling loads (in the order of 1 ton) could be handled by shaft-drive air-cycle units and larger loads by hydraulically driven units.

One variation of this method is shown in fig D-20. It represents basically the elimination of the transport loop. The refrigerant makes direct contact with the equipment to be cooled for maximum cooling efficiency. In this arrangement the blower replaces the compressor, the spray nozzle replaces the expansion valve, and the electronic compartment replaces the evaporator in a typical vapor cycle system.

Still another variation of this system is one described in ref. 49 and shown schematically in fig. D-21. The whole vapor cycle is carried out within the box itself without any ducting. The refrigerant (FC-75, in this example) is evaporated by the heat generating component at the bottom of the box, the vapor rising to the cooled cover by diffusion process (mass transfer due to density difference). The vapor condenses at the cover and transforms into liquid droplet form and is returned to the bottom by gravity forces.

In this particular application, SF_6 (sulphur hexafluoride) was used for its high dielectric strength at low temperature (down to -55°C) and vane axial fans were used to promote heat transfer at the cover.

Electric cooling system coupled with environmental control system--The biggest advantage of coupling electrical and electronic cooling with the cabin environmental control system (ECS) is that the main ducting of the air distributing system is already in existence and all that is necessary in order to incorporate electrical cooling is to resize the ECS to include the additional cooling loads and to provide the extra ducting needed to branch off the cool air to the particular locations.

As shown in fig. D-22 the cooling air can be taken either before or after it enters the cabin. The weight penalty in such a modification can be very small as compared to the whole environmental control system.

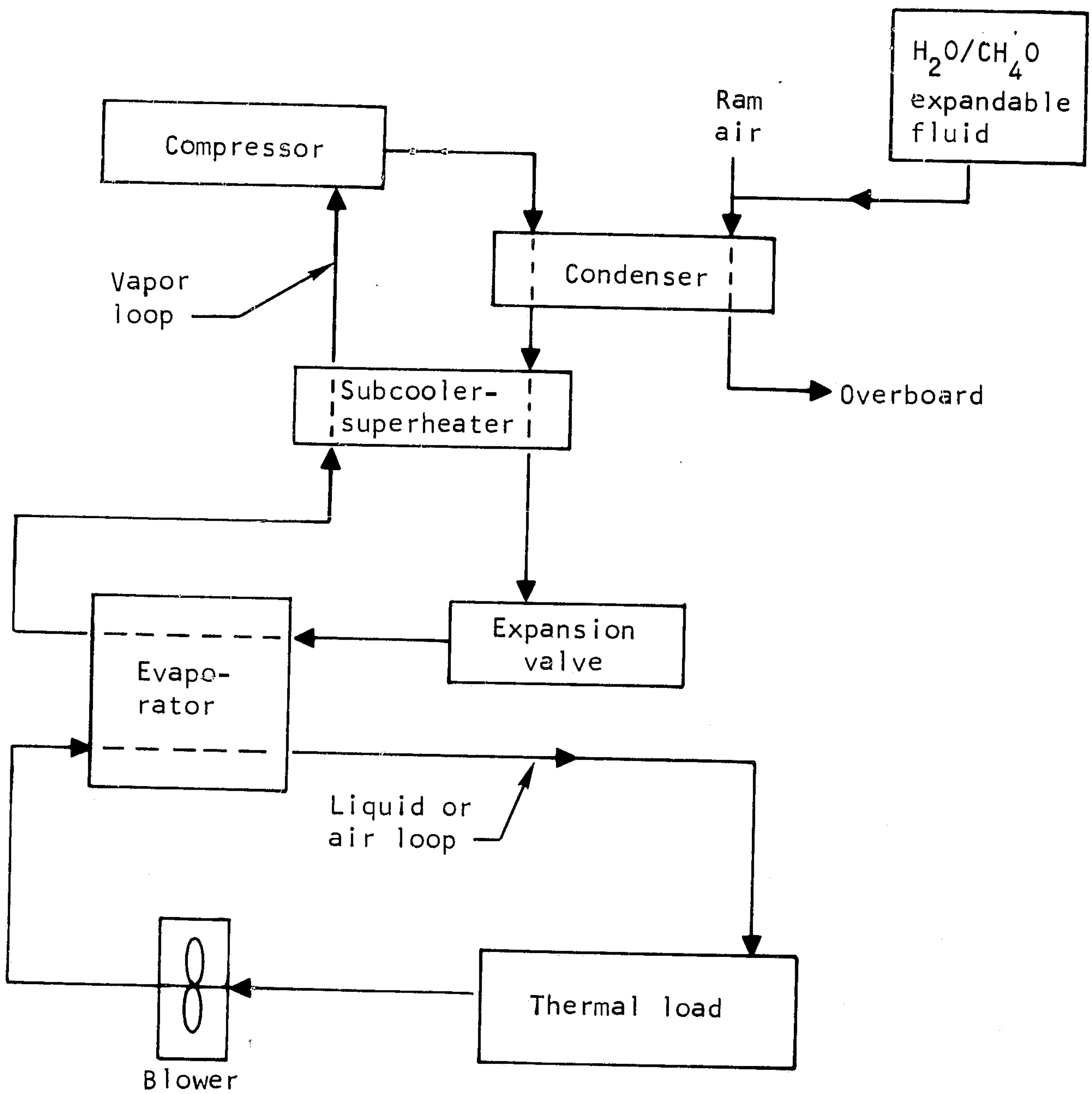

Figure D-19. Vapor Cycle System

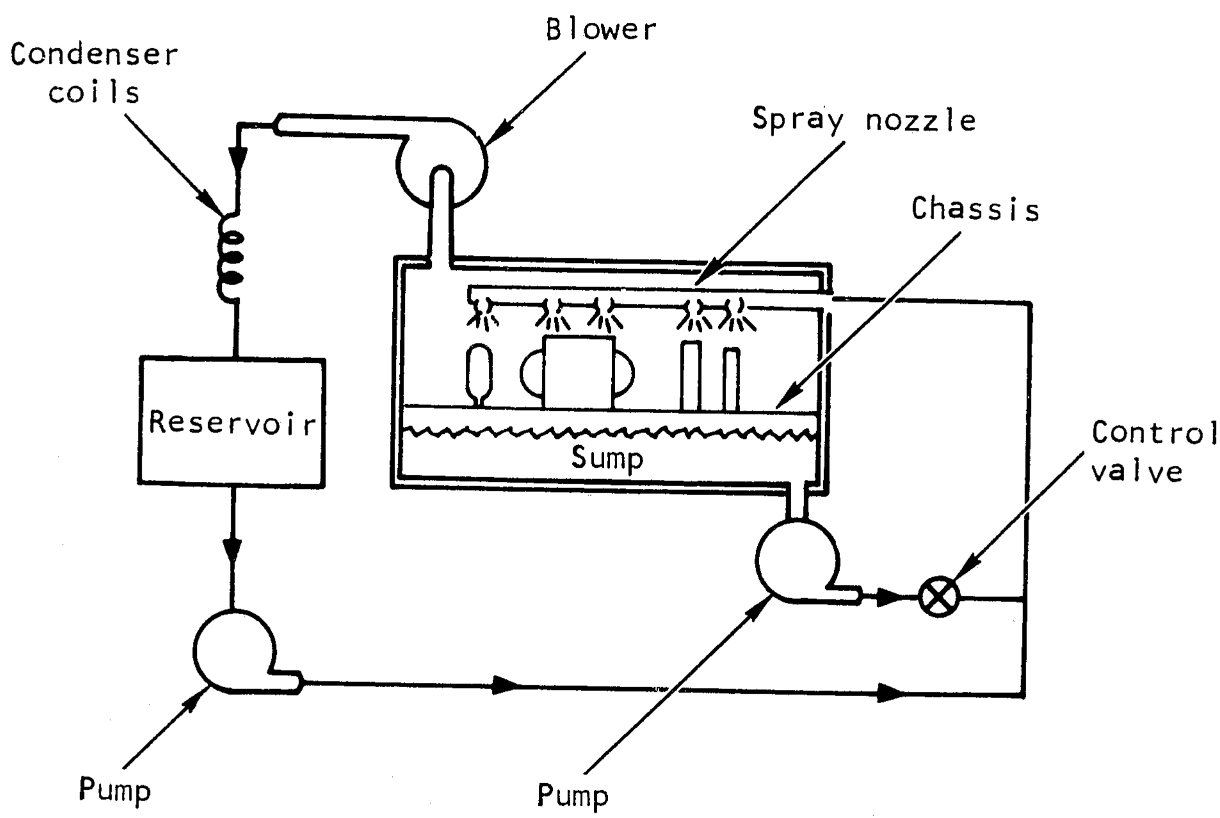

Figure D-20. Electronic Boxes Cooled by Evaporation and Condensation A Direct Evaporative Spray System

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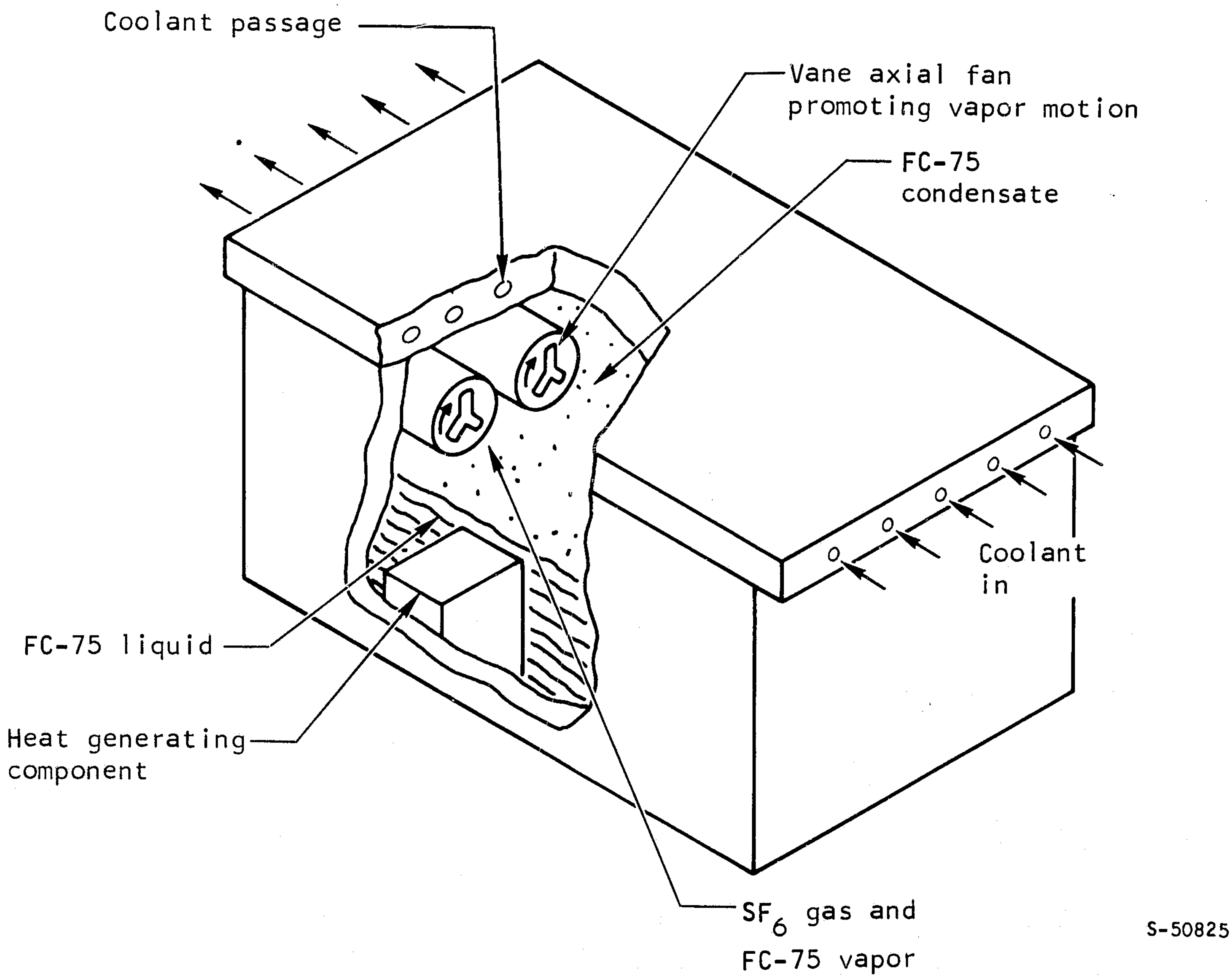

Figure D-21. Evaporation-Condensation Cooling System for High Voltage Power Supply

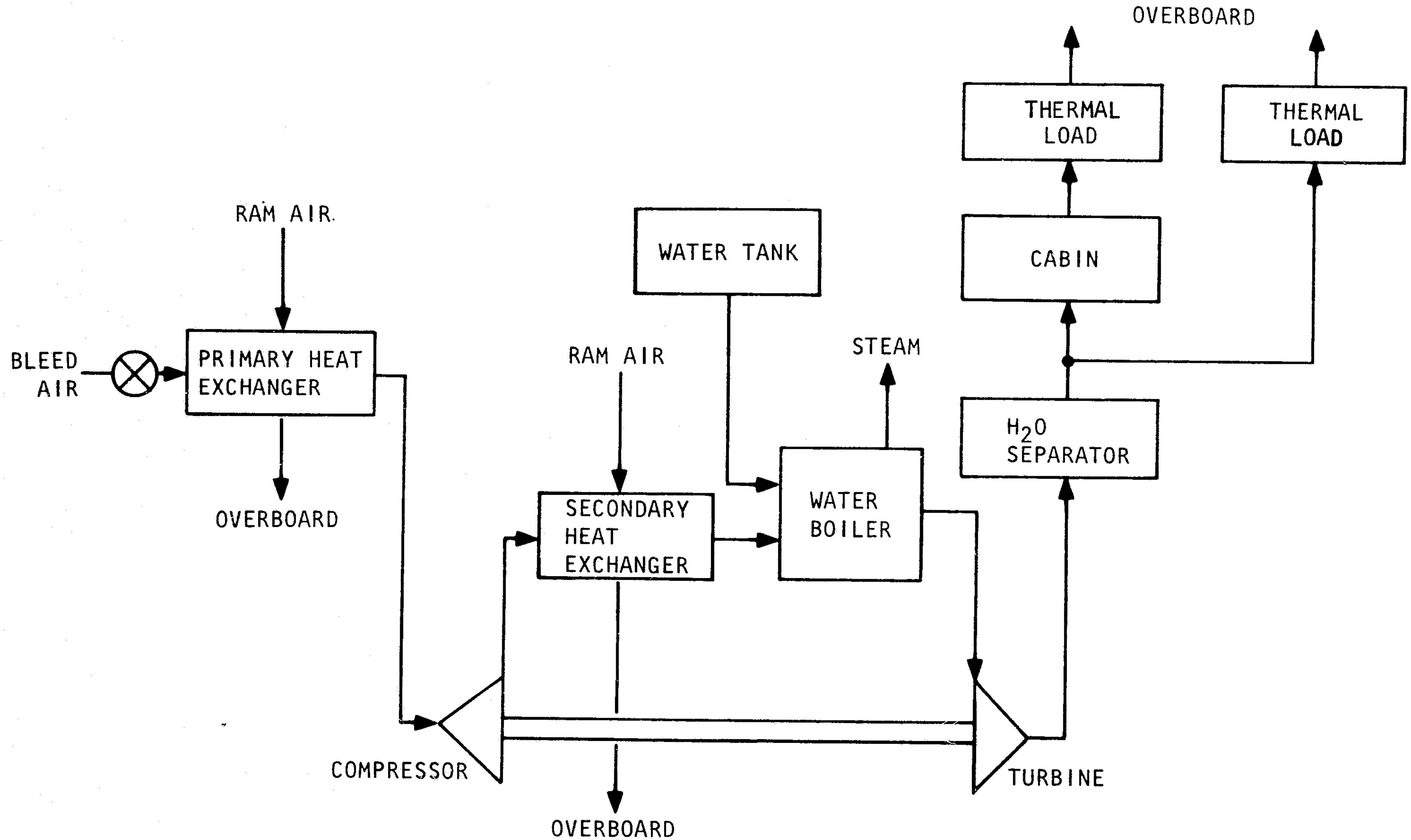

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Figure D-22. The Coupling of Electrical Cooling System with the Environmental Control System (Bootstrap Air Cycle System in this Case)

APPENDIX E

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